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SPACE TRAJECTORIES
ERROR ANALYSIS PROGRAMS
VERSION II

VOLUME II: PROGRAMMER'S MANUAL

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Volume II of Three Volumes

Final Report

Contract NAS 5-11795

Computer Program for Mission Analysis of Lunar and Interplanetary Missions

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FOREWORD

STEAP II is a series of three computer programs developed by the Martin-Marietta Corporation for the mathematical analysis of the navigation and guidance of lunar and interplanetary trajectories. STEAP is an acronym for Space Trajectory Error Analysis Programs. The first series of programs under this name was developed under contract NAS 1-8745 for Langley Research Center and was documented in two volumes (STEAP User's Manual, STEAP Analytical Manual) as NASA Contract Report 66818. Under contract NAS 5-11795 the STEAP series was extensively modified and expanded for Goddard Space Flight Center. This second generation series of programs is referred to as STEAP II.

STEAP II is composed of three independent yet related programs: NOMNAL, ERRAN, and SIMUL. All three programs require the integration of n-body trajectories for both interplanetary and lunar missions. The virtual mass technique is the scheme used for this purpose in all three programs.

The first program named NOMNAL is responsible for the generation of n-body nominal trajectories (either lunar or interplanetary) performing a number of deterministic guidance events. These events include initial or injection targeting, midcourse retargeting, and orbit insertion. A variety of target parameters are available for the targeting events. The actual targeting is done iteratively either by a modified Newton-Raphson algorithm or by a steepest descent-conjugate gradient scheme. Planar and nonplanar strategies are available for the orbit insertion computation. All maneuvers may be executed either by a simple impulsive model or by a pulsing sequence model.

ERRAN, the second program of STEAP II, is used to conduct linear error analysis studies along specific targeted trajectories. The targeted trajectory may however be altered during flight by retargeting events (computed either by linear or nonlinear guidance) and by an orbit insertion event. Knowledge and control covariances are propagated along the trajectory through a series of measurements and guidance events in a totally integrated fashion. The knowledge covariance is processed through measurements using an optimal Kalman-Schmidt filter with arbitrary solve-for/consider augmentation. Execution errors at guidance events may be modeled either by an impulsive approximation or by a pulsing sequence model. The resulting knowledge and control covariances may be analysed by the program at various events to determine statistical data including probabilistic midcourse correction sizing and effectiveness, probability of impact, and biased aimpoint requirements.

The third and final program in the STEAP II series is the simulation program SIMUL. SIMUL is responsible for the testing of the mathematical models used in the navigation and guidance process. An "actual" dynamic model is used to propagate an "actual" trajectory. Noisy measurements from this "actual" trajectory are then sent to the estimation algorithm. Here the actual measurement, the statistics associated with that measurement, and an "assumed" dynamical model are blended together to generate the filter estimate of the trajectory state. This process is repeated continually through the measurement schedule. At guidance events corrections are computed

based on the estimate of the current state. These corrections are then corrupted by execution errors and added to the "actual" trajectory. The statistics and augmentation of the filter, the mismatches in the "actual" and "assumed" dynamics, and the execution errors and measurement biases may then be varied to determine the effects of these parameters on the navigation and guidance process.

The documentation for STEAP II consists of three volumes: the Analytic, Programmer's and User's Manuals. Each of these documents is self-contained.

The Analytic Manual consists of two major divisions. The first section provides a unified treatment of the mathematical analysis of the STEAP II programs. The general problem description, formulation, and solution are given in a tutorial manner. The second section of this report supplies the detailed analysis of those subroutines of STEAP II dealing with technical tasks.

The Programmer's Manual provides the reader with the information he needs to effectively modify the programs. Both the overall structure of the programs as well as the computational flow and analysis of the individual subroutines is described in this manual.

The User's Manual contains the information necessary to operate the programs. The input and output quantities of the programs are described in detail. Example cases are also given and discussed.

CONTENTS

1. INTRODUCTION

This Programmer's Manual is intended to supply the reader with sufficient information about the STEAP II programs to enable him to efficiently modify them. Both the overall structure of the programs and the computational flow of the individual subroutines are described in this manual.

This section describes the contents of the Programmer's Manual. Following this discussion the nomenclature used throughout the report is presented.

The third section of this manual describes the four basic components of STEAP II: the n-body trajectory propagation package, the nominal trajectory generator NOMNAL, the error analysis program ERRAN, and the simulation program SIMUL. The general purpose and capability of each of the programs is briefly summarized.

Chapter 4 of this volume examines the STEAP II programs from a more detailed viewpoint. The operational structure of each of the main components is described at the subroutine level. The individual subroutines are defined and cross-referenced according to the three main programs of STEAP II.

Chapter 5 contains the definitions of the variables appearing in common blocks throughout the programs. The variables are first listed according to the common blocks to which they belong. The programs requiring each of these common blocks are also noted. Following this all the common variables are listed in alphabetical order with their common blocks referenced. Tables detailing the definitions of large, frequently referenced common arrays are also provided.

Chapter 6 comprises the bulk of this volume. Each of the subroutines is documented in detail in alphabetical order. The purpose of the subroutine is supplied. Subroutines supported or required by the subroutine are listed. Arguments and interval variables of the subroutine are defined and usage of common variables is noted. The mathematical analysis upon which the subroutine is based is then discussed in full. Finally a flow chart of the computational flow of the subroutine is provided.

2. NOMENCLATURE

A. Arabic symbols

Symbol	Definition
a	Semi-major axis of conic
$B \cdot T$	Impact plane parameter
$B \cdot R$	Impact plane parameter
C_{xx_s}	Correlation between position/velocity state and solve-for parameters
C_{xu}	Correlation between position/velocity state and dynamic consider parameters
C_{xv}	Correlation between position/velocity state and measurement consider parameters
$C_{x_s u}$	Correlation between solve-for parameters and dynamic consider parameters
$C_{x_s v}$	Correlation between solve-for parameters and measurement consider parameters
e	Eccentricity of conic
E	Eccentric anomaly
f	True anomaly on conic
G	Observation matrix relating observables to dynamic consider parameter state
H	Observation matrix relating observables to position/velocity state
i	Inclination of conic (reference body equatorial)
J	Measurement residual covariance matrix
K	Kalman gain constant for position/velocity state
L	Observation matrix relating observables to measurement consider parameter state Mean longitude
M	Observation matrix relating observables to solve-for parameter state Mean anomaly

n_1	Dimension of solve-for parameter state
n_2	Dimension of dynamic consider parameter state
n_3	Dimension of measurement consider parameter state
p	Semilatus rectum of conic Probability density function
P	Position/velocity covariance matrix
$\hat{P}$	Unit vector to periapsis of conic
P_s	Solve-for parameter covariance matrix
Q	Dynamic noise covariance matrix
$\tilde{Q}$	Execution error matrix
$\hat{Q}$	Unit vector in plane of motion normal to P
r	Radius
r_{CA}	Radius of closest approach
r_{SI}	Radius of sphère of influence
R	Measurement noise covariance matrix
$\underline{R}$	Actual noise covariance matrix
$\hat{R}$	Unit vector normal to T in plane perpendicular to approach asymptote directed south ($R = S \times T$)
R_c	Target planet capture radius
S	Kalman gain constant for solve-for parameters
S_j	Velocity correction covariance matrix
$\hat{S}$	Approach or departure asymptote
t_{CA}	Time of closest approach to target body
t_{SI}	Time of intersection with sphere of influence of target body
Δt	Time interval
$\hat{T}$	Unit vector lying in ecliptic plane normal to $\hat{S}$. ($\hat{T} = \frac{\hat{S} \times \hat{K}}{\lvert \hat{S} \times \hat{K} \rvert}$ where $\hat{K}$ is unit normal to ecliptic plane.)
U_o	Dynamic consider parameter covariance matrix

v	Velocity
V_o	Measurement consider parameter covariance matrix
W_j	Target parameter covariance matrix
$\hat{W}$	Unit normal to orbital plane
X	Actual position/velocity state
$\bar{X}$	Targeted nominal position/velocity state
$\tilde{X}$	Most recent nominal position/velocity state

B. Greek Symbols

α	Auxiliary parameters
Γ_j	Guidance matrix
Γ	Flight path angle
δ	Declination of vector
Δv	Velocity increment
ϵ	Measurement residual Errors in target parameters
η_j	Variation matrix relating position/velocity variations to target conditions
θ_{xx_s}	State transition matrix partition associated with solve-for parameters
θ_{xu}	State transition metrix partition associated with dynamic consider parameters
θ	Longitude or right ascension
Λ_j	Projection of target condition covariance matrix W_j into the impact plane
μ	Gravitational constant of body
$\vec{\mu}$	Biased aimpoint
ν	Sampled measurement noise True anomaly

ρ	Magnitude of gaussian approximation for midcourse correction Correlation coefficient
σ	Standard deviation
Σ	Launch azimuth
$\vec{T}$	Target parameters
Φ	Targeting matrix State transition matrix for position/velocity state Latitude
χ	Sensitivity matrix
ψ_j	Matrix relating guidance corrections to target condition deviations
Ω	Longitude of ascending node
ω	Argument of periapsis
$\tilde{\omega}$	Longitude of periapsis

C. Subscripts

C	Control variable (P_C)
CA	Closest approach (r_{CA})
f	Final variable (t_f)
i	Initial variable (t_i)
j	Index of current guidance event (P_j)
k	Index of current measurement (P_k)
K	Knowledge variable (P_K)
s	Solve-for parameter (x_s)
SI	Sphere of influence (t_{SI})

D. Superscripts

A	Augmented variable (Φ^A)
T	Matrix transpose (Φ^T)
-1	Matrix inverse (Φ^{-1})

$-$	Variable immediately before instant (P_k^- or v^-)
$+$	Variable immediately after instant (P_k^+ or v^+)

E. Abbreviations

AU	Astronomical unit
CA	Closest approach to reference body
ERRAN	Error analysis program
FTA	Fixed time of arrival guidance policy
GHA	Greenwich hour angle
J.D.	Julian date (referenced either 0^{yr} or 1900^{yr})
km	Kilometers
M/C	Midcourse correction
NOMNAL	Nominal trajectory generation program
POI	Probability of impact
Q-L	Quasilinear filter event
S/C	Spacecraft
SF/C	Solve-for/consider
SIMUL	Simulation program
SOI	Sphere of influence
STM	State transition matrix
STEAP	Space Trajectories Error Analysis Programs
VM	Virtual Mass
2VBP	Two variable B-plane guidance policy
3VBP	Three variable B-plane guidance policy

3. SUMMARY OF MODES

The Space Trajectory Error Analysis Programs (STEAP) consist of four subprograms or operational modes. The first mode, used as a subroutine by each of the other three programs, is the trajectory mode VMP by which an n-body trajectory (lunar or interplanetary) is propagated by the virtual mass technique. The second mode is the nominal trajectory generator or targeter (NOMNAL) by which a lunar or interplanetary trajectory meeting specified conditions is determined. The third mode is the error analysis program ERRAN in which the navigation and guidance characteristics of a nominal trajectory are analyzed by linearly propagating knowledge and control covariances along the trajectory. Finally the simulation mode SIMUL tests the mathematical models used in the navigation and guidance processes by modeling the tracking and correction of an "actual" trajectory. In this chapter a general description of each of these modes will be provided.

3.1 The Virtual Mass Propagator VMP

The dynamic model used by STEAP is supplied by the trajectory propagation package. The only external forces acting upon the spacecraft are assumed to be the gravitational forces of the celestial bodies considered in the integration. Both the spacecraft and the gravitational bodies are assumed to be point masses so neither spacecraft attitude nor planet asphericities are considered.

The celestial bodies to be in the integration are specified by the user and may include the sun, any of the nine planets, and the earth's moon. The motion of the planets about the sun and the moon about the earth are modeled by using mean ecliptic elements of date. If the user desires, each of the planets can be set in a fixed ellipse referenced to some epoch for speedier computation.

The coordinate system used in the integration is also specified by the user. The options available are either heliocentric ecliptic or barycentric ecliptic (nominally for lunar trajectories).

The actual scheme used in the propagation of the trajectory is the virtual mass or yaricentric technique (see reference 15). No actual integration is performed by the trajectory mode; the key idea of the virtual mass technique is to build up an n-body trajectory by using a sequence of conic sections around a moving effective force center called the virtual mass. At each instantaneous moment along the trajectory, the combined effects of all the gravitational bodies can be viewed as resulting from a fictitious body of unique magnitude and position which is called the virtual mass. The computational pro-

cedure then assumes that over a small time interval the motion of the spacecraft can be represented by a two-body conic section arc relative to this virtual mass. The complete trajectory is thus generated by a series of small arcs pieced together in steps while updating the position and magnitude of the effective force center. The main advantage of the virtual mass technique is that the tedious numerical integration of the differential equations is avoided.

Another significant feature of the virtual mass technique is its flexibility. By varying a simple parameter called the "accuracy level" related to the true anomaly increment of each step, trajectories ranging from a sequence of relatively few conic section arcs corresponding to a very approximate solution to those requiring a large number of arcs corresponding to highly accurate solutions may be generated.

3.2 The Nominal Trajectory Targeter NOMNAL

NOMNAL is responsible for the generation of a nominal trajectory for either lunar or interplanetary missions. The method of propagation in either case is the virtual mass n-body integrator. The trajectory may be processed through a series of deterministic maneuvers including initial or injection targeting, subsequent retargeting, and finally orbit insertion. A variety of target parameters are available for the targeting events. Both coplanar and nonplanar strategies are permitted in the orbit insertion maneuver.

If an initial state for the problem is known, this may be read in to start the trajectory. Otherwise NOMNAL generates its own zero iterate. In interplanetary missions this involves solving the Lambert time of flight equation for the massless planet trajectory that connects the desired initial and final positions in the specified time interval. Four options are available in describing these reference points:

Initial Point	Final Point
Launch Planet	Target Planet
Launch Planet	Specified Point
Specified Point	Target Planet
Specified Point	Specified Point

If the initial point is referenced to the launch planet, a launch profile is consulted to generate a realistic set of injection condition consistent with the heliocentric trajectory.

For lunar trajectories a slightly different procedure is used. The required data for the lunar zero iterate includes specification of the desired semimajor axis with respect to the moon, radius and time of

closest approach to the moon, and inclination to the lunar equator. Then the generation of the zero iterate is accomplished by first targeting a patched conic trajectory and then a multi-conic trajectory to the desired conditions.

A targeting event may be processed immediately after obtaining a zero iterate state or at any point along the nominal trajectory. At a targeting event the current velocity is refined to yield a trajectory satisfying target parameter constraints. The possible target parameters are:

1.	TRF	5.	B·T	9.	SMA (Lunar)
2.	TSI	6.	B·R	10.	XF
3.	TCS	7.	RCA	11.	YF
4.	TCA	8.	INC	12.	ZF

The targeting method to be used is specified by the user. Either a modified Newton-Raphson algorithm or a steepest descent/conjugate gradient technique may be used.

Orbit insertion events are also available in NOMNAL. At a specified time the spacecraft state relative to the target body is computed. The resulting conic trajectory relative to the target body is then compared with the desired orbit to determine the optimal time to make the insertion and the required correction. At the proper time the velocity correction is then implemented. Two strategies are permitted in the orbit insertion computation:

Coplanar - The desired semimajor axis, eccentricity, and periapsis shift of a coplanar orbit are specified.

Noncoplanar - The desired plane of the post-insertion state is specified along with nominal values of the orbit elements.

The targeted correction, orbit insertion correction, or an externally supplied correction may be executed if desired. Two models are available for this implementation; a simple impulsive addition or a more complex multiple pulse model.

The program will integrate and record the periodically-corrected nominal trajectory until reaching a termination time specified by the user.

3.3 The Error Analysis Program ERRAN

The error analysis program ERRAN is a preflight mission analysis tool used primarily to propagate covariance matrices along selected

The guidance event is the most complex event and yields much useful information for preflight mission analysis. Several types of guidance events are available in ERRAN. At a midcourse guidance event the user can choose from three midcourse guidance policies. The midcourse guidance event can also be constrained to satisfy planetary quarantine requirements. At an orbital insertion guidance event the user can choose from two insertion policies. Options are also available for changing target conditions in mid-flight and retargeting the trajectory using nonlinear techniques, or for simply applying an externally-supplied or precomputed ΔV at some arbitrary trajectory time. Two thrust models are available: impulse and impulse series. Execution error statistics are generated using an error model defined by a proportionality error, a resolution error, and two pointing angle errors. At a midcourse guidance event in ERRAN we also compute a statistical ΔV and the target condition covariance matrix both before and after the midcourse correction.

3.4 The Simulation Program SIMUL

The simulation program SIMUL is the most complex program in the STEAP set of programs. In SIMUL the validity of the navigation and guidance process is examined by simulating an actual mission. Spacecraft state estimates are generated in SIMUL, as well as covariance matrices. The results given by the error analysis program ERRAN become meaningful only when SIMUL shows that the estimated spacecraft trajectory converges, within reasonable bounds specified by the covariance matrix, to the simulated actual trajectory.

All state transition matrix, parameter augmentation, and measurement options described in section 3.3 are also available in SIMUL. As in ERRAN, the computational procedure in SIMUL is divided into basic cycle computations and event computations. The SIMUL basic cycle is concerned with the generation of state estimates and an actual trajectory, together with all quantities generated in the ERRAN basic cycle. Eigenvector and prediction events in SIMUL involve all computations performed in the corresponding ERRAN events. In addition, the SIMUL prediction event propagates state estimates forward to the time to which we are predicting.

All options available in the ERRAN guidance event (see section 3.3) are also available in the SIMUL guidance event. The treatment of the midcourse guidance event, however, is different in several respects. First, since an estimated spacecraft state is generated in SIMUL, an actual midcourse ΔV can be computed, rather than a statistical ΔV as in ERRAN. Also, all linear midcourse ΔV's computed in SIMUL can be recomputed using nonlinear techniques. Finally, since an actual trajectory is generated in SIMUL, actual target errors after the midcourse correction are also computed.

interplanetary or lunar trajectories. Three main quantitative results are available from ERRAN: (a) knowledge covariances, which provide a measure of how well the actual trajectory is known after each measurement is processed; (b) control covariances, which when propagated forward to the target provide a measure of how well the nominal target conditions will be satisfied by the actual trajectory; and (c) statistical midcourse ΔV's.

State transition matrices are required to propagate covariance matrices over an arbitrary interval of time. Three methods are available for computing the 6 x 6 position/velocity state transition matrix. The first two methods, which are analytical methods, are analytical patched conic and analytical virtual mass. The third method uses numerical differencing to compute the state transition matrix. To increase the accuracy of the analytical techniques over long time intervals a state transition matrix cascading option is also available. Augmented parameter state transition matrices are always computed using numerical differencing.

Measurements are processed in an optimal recursive consider filter. Up to 23 dynamic and measurement parameters may be solved-for or considered. The dynamic parameters include biases in the gravitational constants of the Sun and the target planet and biases in the 6 orbital elements of the target planet. Measurement biases include biases in the locations of the 3 earth-based tracking stations, and biases in all measurements. Available measurement types are range, range-rate, star-planet angles, and apparent planet diameter measurements. Measurement noise for each measurement type is assumed to be constant.

The computational procedure in ERRAN is divided into basic cycle computations and event computations. Basic cycle computations are concerned with the propagation of covariances forward to a measurement time and processing the measurement. Events refer to a set of specialized computation, not directly concerned with measurement processing, which can be scheduled to occur at arbitrary times along the trajectory.

The three events available in ERRAN are eigenvector events, prediction events, and guidance events. At an eigenvector event the position and velocity partitions of the knowledge covariance matrix are diagonalized to reveal geometric information about the size and orientation of the position and velocity navigation uncertainties. Associated hyperellipsoids are also computed. At a prediction event the most recent covariance matrix is propagated forward to some critical trajectory time to determine predicted navigation uncertainties in the absence of further measurements.

A quasi-linear filtering event, not defined in ERRAN, is also available in SIMUL. At a quasi-linear filtering event the most recent nominal trajectory is updated by using the most recent state estimate. This permits more accurate computation of state transition and observation matrices which in turn helps prevent the occurrence of divergence of the state estimate.

4. DESCRIPTION OF SUBROUTINES

4.1 Index of Subroutines

The subroutines making up the STEAP programs are listed according to category in Table 4.1 following. The programs are divided into three general classes: the subroutines making up the virtual mass propagation package used by the three basic programs, the additional subroutines required by NOMNAL and then the additional subroutines used in ERRAN and SIMUL. In Table 4.2 the subroutines are listed again by category with a brief summary of their purpose. Thus Table 4.2 can be used to track down the subroutine in which a specific task is performed. The individual subroutines are then documented in detail in alphabetical order in Chapter 6.

4.2 VMP Subroutine Hierarchy

The executive program for the virtual mass n-body trajectory propagator is named VMP. The reader should investigate the detailed analysis and flow chart of VMP in the individual subroutine documentation in Chapter 6. The summaries of the subroutines of VMP are given in the first part of Table 4.2. The subroutines are conveniently divided into four general classes:

Conic	Subroutines based on conic approximations
Ephemeris	Subroutines used to compute the positions and velocities of the gravitational bodies at different times along the trajectory
Propagation	Subroutines used in the direct computation of the trajectory of the spacecraft moving under the influence of all the gravitational bodies
Input/Output	Subroutines processing either the input or output from the virtual mass trajectory propagation

The calling hierarchy of the virtual mass programs is given in Figure 4.1. All subroutines within a given block are at an identical level relative to the calling hierarchy unless they are enclosed by parentheses. Subroutines within parentheses are called by the preceding subroutine. Otherwise calls to subroutines are indicated by arrows. Thus all subroutines within blocks connected directly to VMP are called directly from VMP.

4.3 NOMNAL Subroutine Hierarchy

The first of the three independent programs of STEAP is the nominal trajectory targeter NOMNAL. The main program controlling the processing of the program goes under the same name. Reference is made to the complete documentation of NOMNAL in Chapter 6. The subroutine hierarchy of NOMNAL is provided in Figure 4.2. BLOCK DATA loads the planetary constants used by many of the subroutines; it is therefore available to all subroutines of NOMNAL. PRELIM reads the input data and calls ZERIT for the computation of a zero iterate if necessary. ZERIT in turn calls HELIO or LUNA for the actual computation of the interplanetary or lunar zero iterate respectively. NOMNAL calls TRJTRY for the propagation of the nominal trajectory between guidance maneuvers. TRJTRY of course calls the VMP package described in Figure 4.1. NOMNAL calls GIDANS for the actual processing of any guidance event. GIDANS calls VMP to initialize arrays for the other events. If a targeting event requires a zero iterate computation ZERIT is called. Subroutine TARGET controls the targeting events; INSERS controls the insertion decision computations. NOMNAL calls EXCUTE for the execution of either type event.

4.4 ERRAN and SIMUL Hierarchy

The calling hierarchy of the subroutines used in ERRAN and SIMUL is shown in Figures 4.3 and 4.4, respectively. The similar structure of ERRAN and SIMUL is apparent from these two figures. All subroutines can be classified under one or more of the following categories: input, output, basic cycle (measurement processing), or events.

The calling hierarchy of the subroutines is indicated by the level of the subroutine in figures 4.3 and 4.4. A given subroutine calls all those subroutines which are directly connected to the subroutine and are located on the next lower level. For the purposes of clarity, the lowest level subroutine on a given branch is enclosed in parentheses. BLOCK DATA is shown connected to the main hierarchy with a dashed line to indicate that the constants stored in BLOCK DATA are available to all subroutines.

The complete documentation of all subroutines used in ERRAN and SIMUL is given in Chapter 5 of this document.

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Table 4.1 STEAP II Subroutines

I. Virtual Mass Subroutines

A. Conic
1. CAREL
2. ELCAR
3. IMPACT

B. Ephemeris
1. TIME
2. BLOCK DATA
3. ORB
4. EPHEM
5. CENTER
6. PECEQ
7. EULMX

C. Propagation
1. VMP
2. ESTMT
3. VECTOR
4. VMASS

D. Input/Output
1. TRAPAR
2. INPUTZ
3. PRINT
4. SPACE
5. NEWPGE

II. NOMINAL Subroutines

A. Executive
1. NOMNAL
2. PRELIM
3. TRJTRY
4. GIDANS
5. EXCUTE

B. Zero Iterate
1. ZERIT
2. HELIO
3. LAUNCH
4. FLITE
5. SERIE
6. LUNA
7. LUNCON
8. [illegible]NTAR
9. [illegible]ULCON
10. MULTAR
11. BATCON
12. MATIN

C. Targeting
1. TARGET
2. TAROPT
3. TARMAX
4. DESENT
5. MATIN

D. Insertion
1. INSERS
2. COPINS
3. NONINS

E. Pulsing Arc
1. PREPUL
2. PULSEX
3. PERHEL
4. (BATCON)

III. ERRAN and SIMUL Subroutines

A. Executive
1. ERRAN
2. SIMUL

B. Dynamic Model
1. NTM
2. NTMS
3. PSIM
4. NDTM
5. PLND
6. MUND
7. PCTM
8. CONCZ
9. CASCAD

C. Navigation
1. NAVM
2. SCHED
3. TRAKM
4. TRAKS
5. TARPRL
6. STAPRL
7. MENO
8. MENOS
9. BIAS
10. RNUM
11. DYNO
12. DYNOS
13. GHA
14. JACOBI
15. HYELS
16. EIGHY

D. Event
1. SETEVN
2. DETEVS
3. PRED
4. PRESIM
5. QUASI
6. GUIDM
7. GUISIM
8. GUID
9. GUIS
10. VARADA
11. VARSIM
12. PARTL
13. BIAIM
14. POICOM
15. QCOMP
16. NONLIN
17. PULCOV
18. EXCUT
19. EXCUTS

E. Input/Output
1. DATA
2. DATA1
3. DATAS
4. DATA1S
5. CONURT
6. TRANS
7. CORREL
8. STMPR
9. SUB1
10. TITLE
11. PRINT3
12. PRNTS3
13. PRINT4
14. PRNTS4

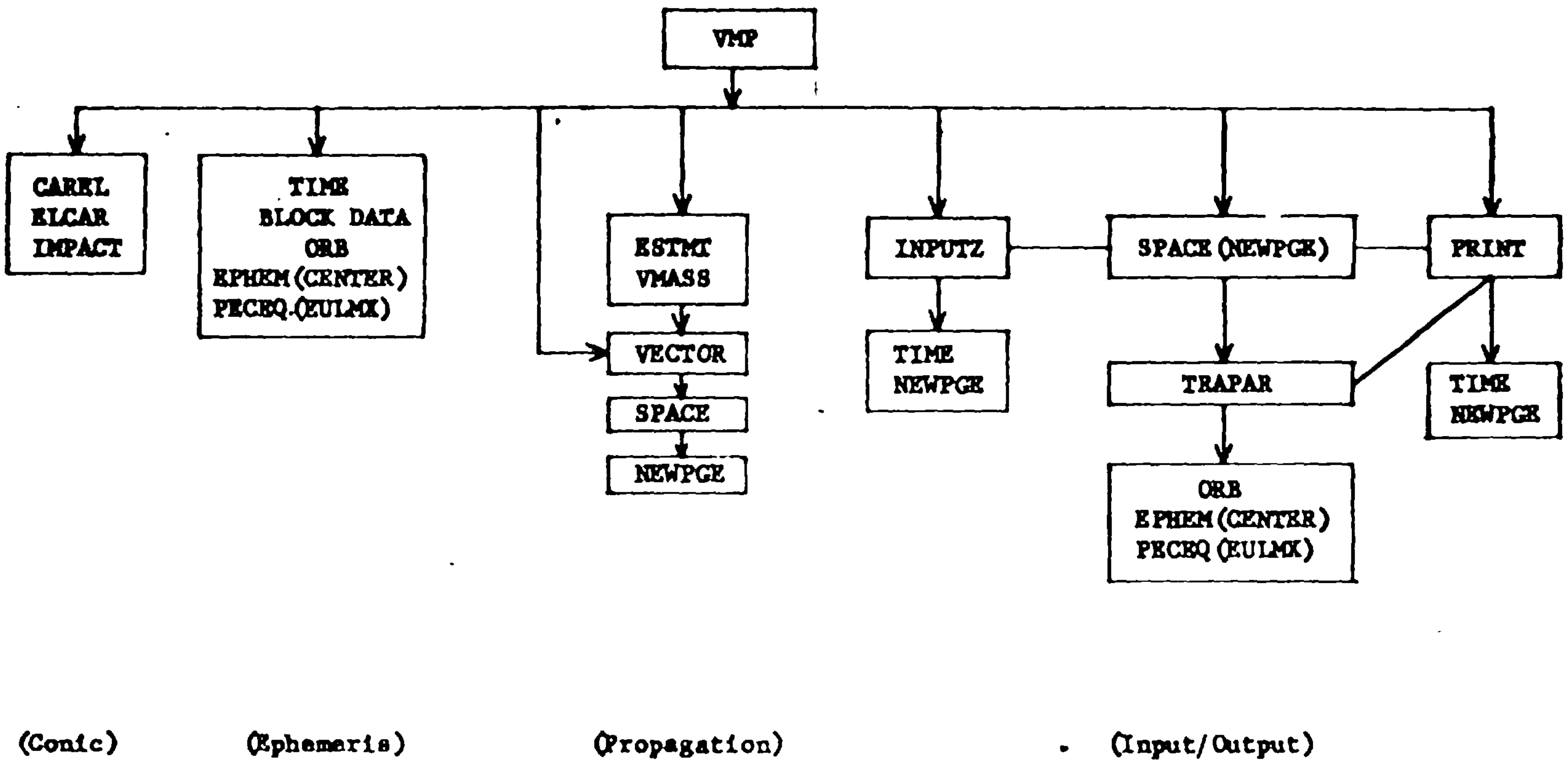
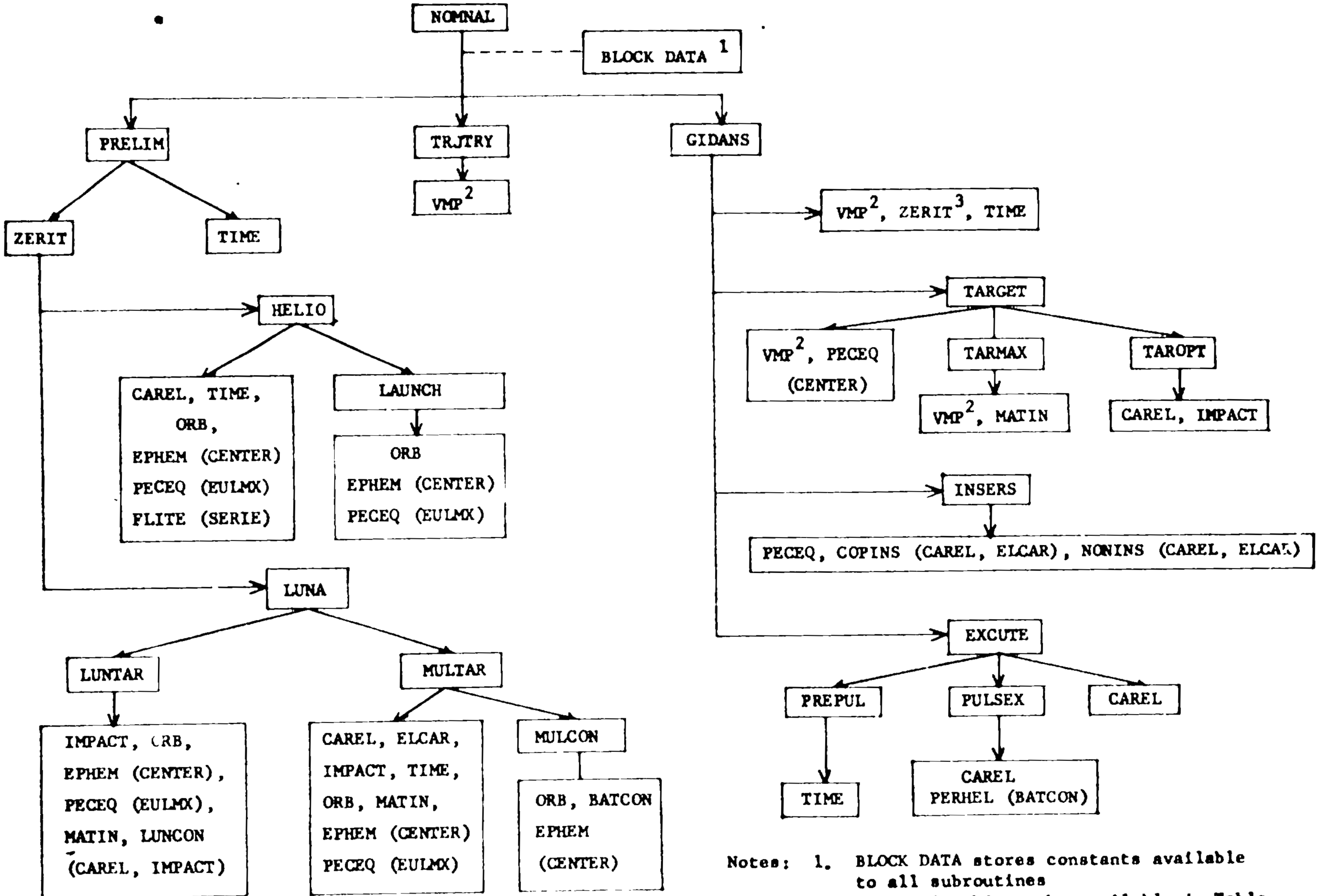
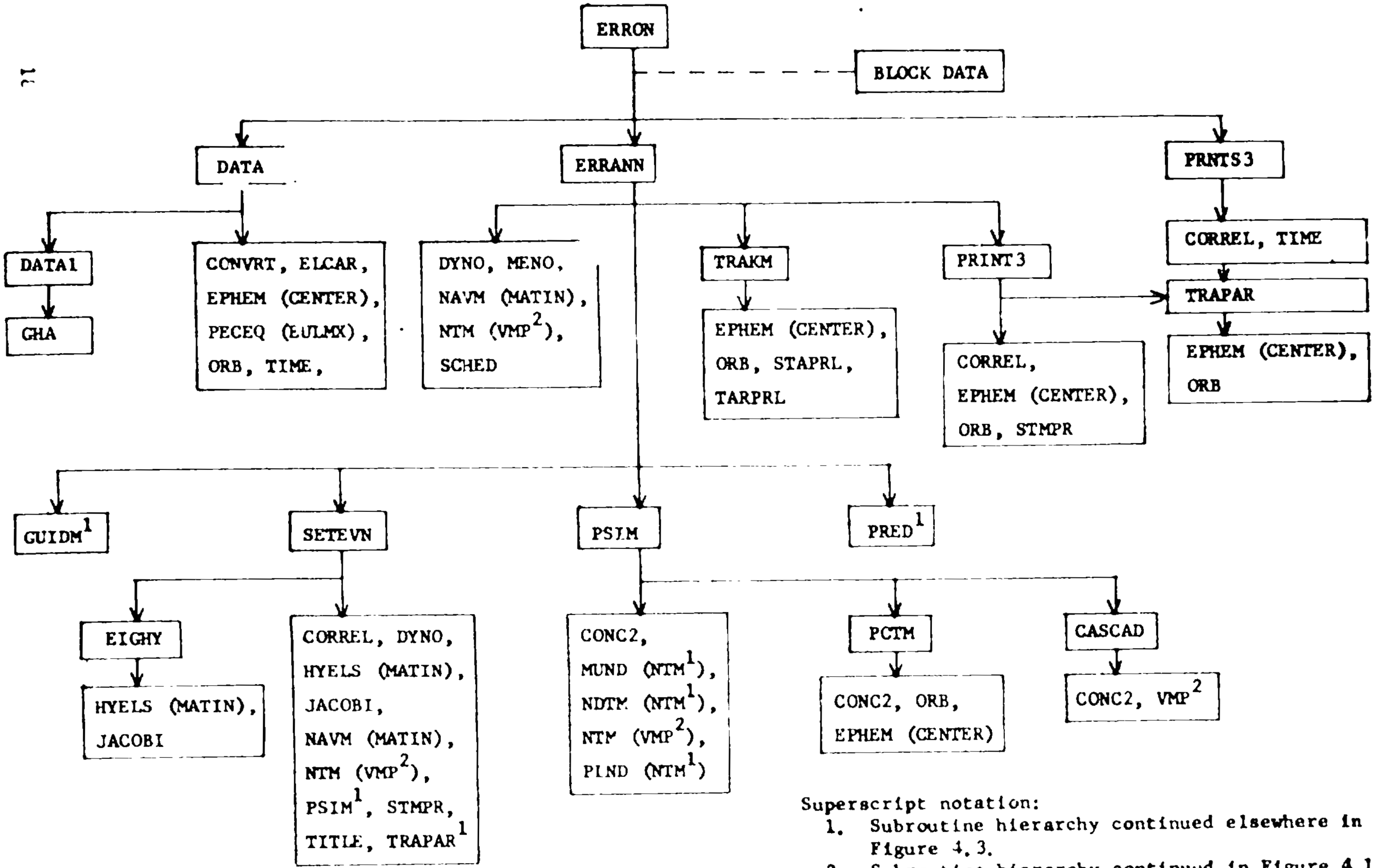


Figure 4.1 Subroutine Hierarchy of VMP



Notes:
1. BLOCK DATA stores constants available to all subroutines
2. Subroutine hierarchy available in Table 4.1
3. Subroutine hierarchy available on far left

Figure 4.2 Subroutine Hierarchy of NOMNAL



Superscript notation:
1. Subroutine hierarchy continued elsewhere in Figure 4.3.
2. Subroutine hierarchy continued in Figure 4.1.
3. Subroutine hierarchy continued in Figure 4.2.

Figure 4.3a Subroutine Hierarchy of ERRAN

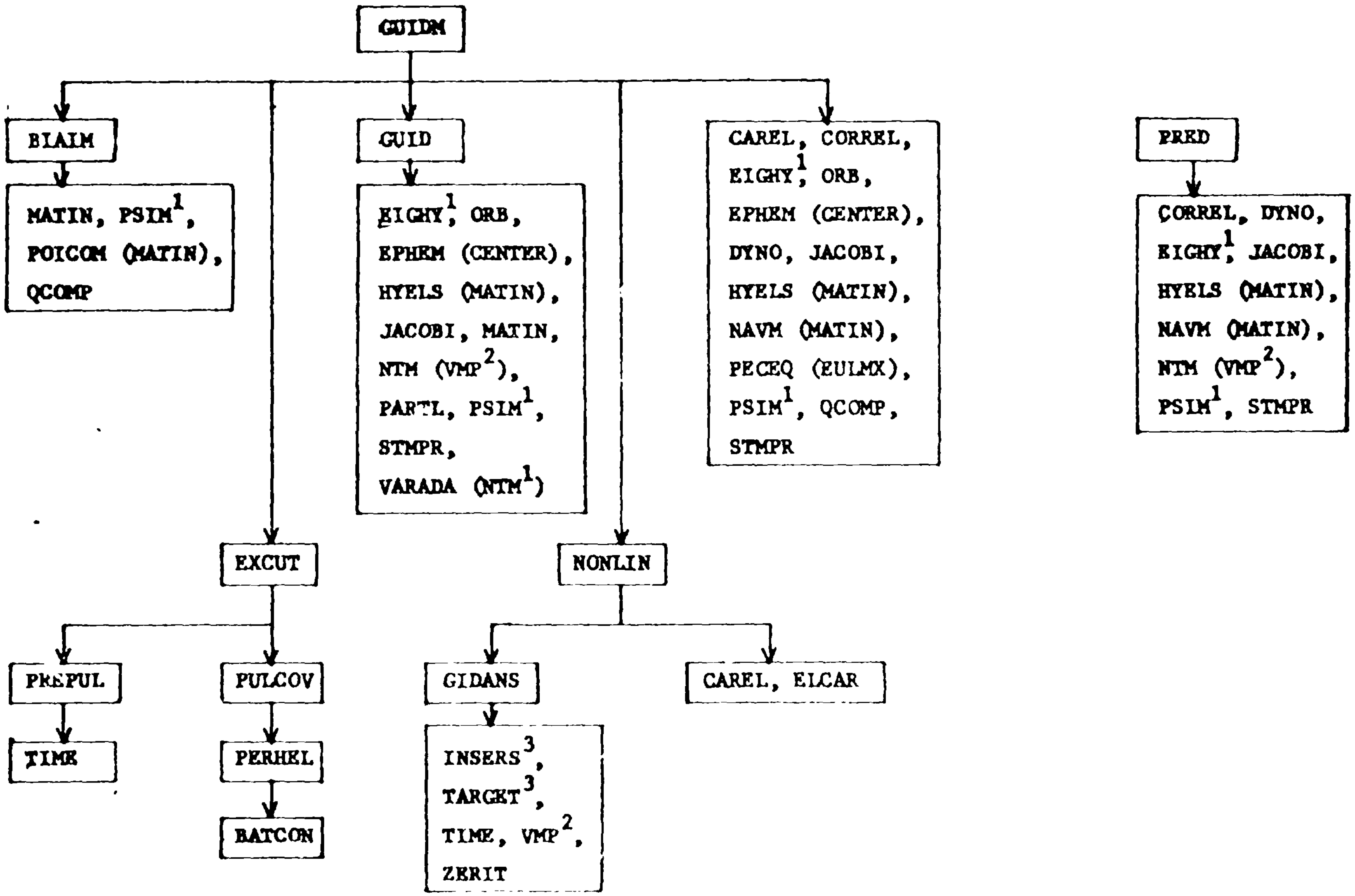
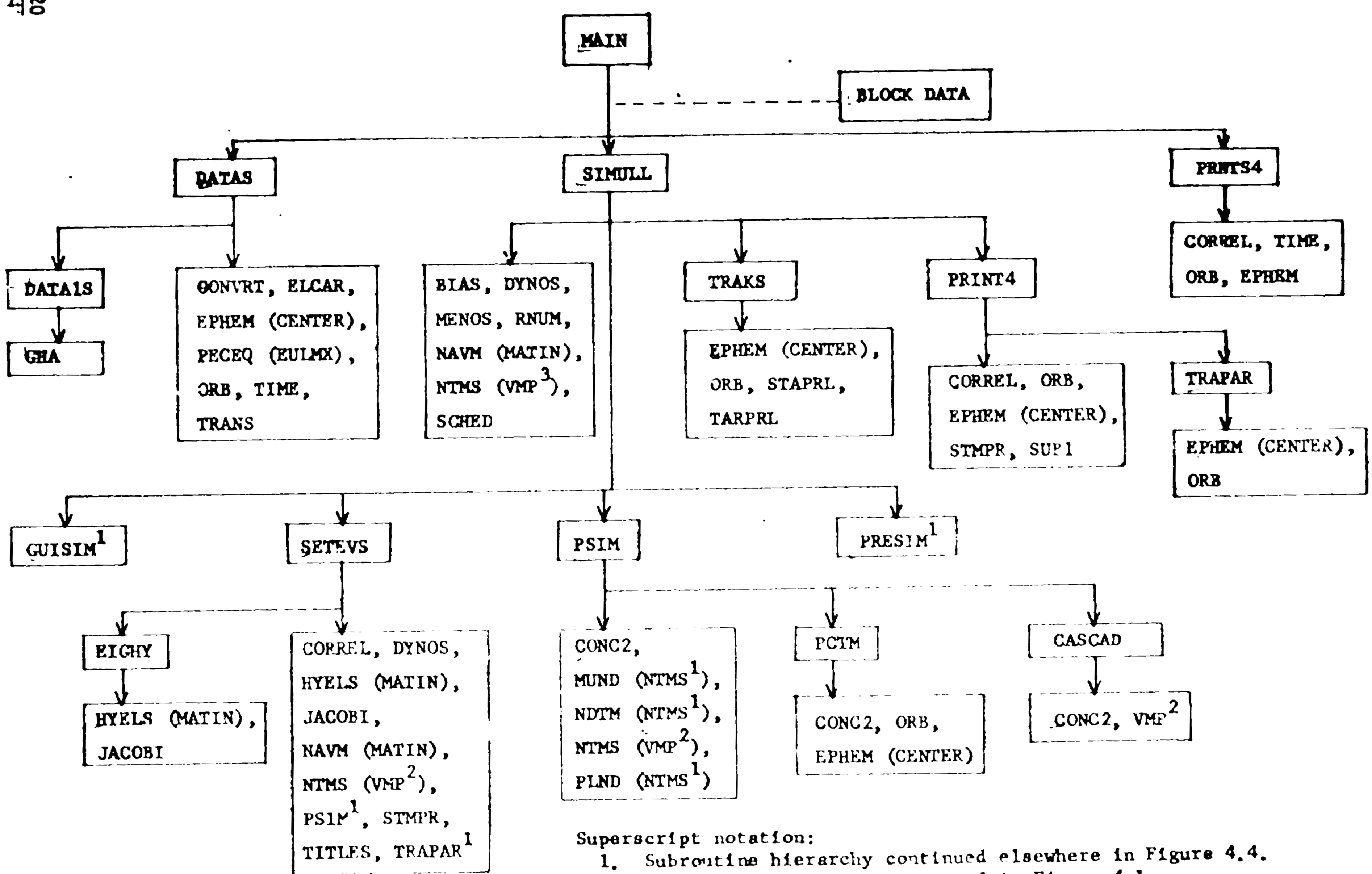


Figure 4.3b Subroutine Hierarchy of ERRAN (continued)



Superscript notation:
1. Subroutine hierarchy continued elsewhere in Figure 4.4.
2. Subroutine hierarchy continued in Figure 4.1.
3. Subroutine hierarchy continued in Figure 4.2.

Figure 4.4a Subroutine Hierarchy of SIMULL

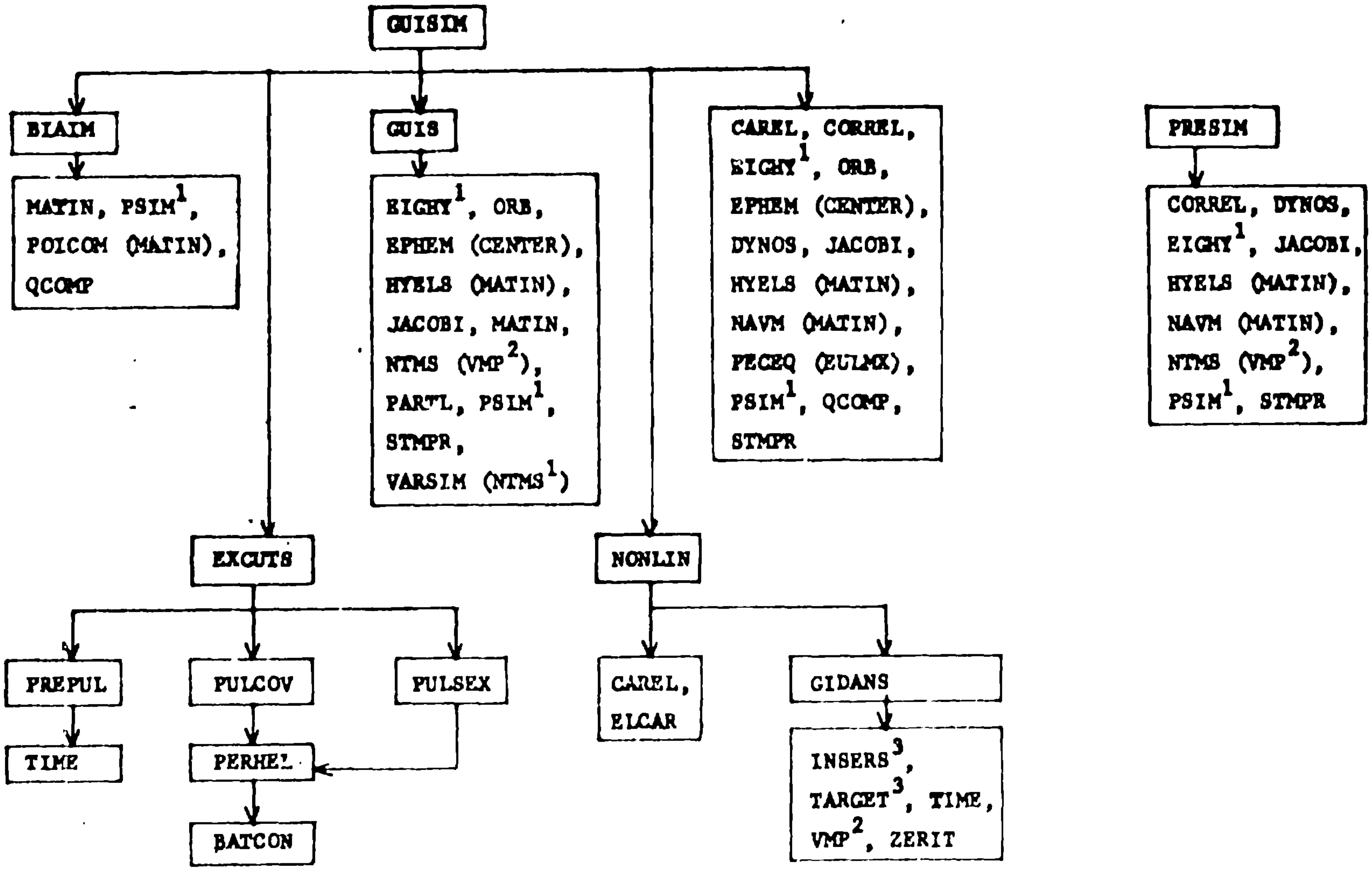


Figure 4.4b Subroutine Hierarchy of SIMUL (continued)

Table 4.2 STEAP II Subroutine Summaries

Subroutine	Function
I. Virtual Mass Subroutines	
A. Conic	
1. CAREL	Convert Cartesian state to conic elements
2. ELCAR	Convert conic elements to Cartesian state
3. IMPACT	Compute impact plane parameters
B. Ephemeris	
1. TIME	Convert J.D. to calendar date or vice versa
2. BLOCK DATA	Set gravitational body ephemeris constants
3. ORB	Compute orbital elements of gravitational body at given time
4. EPHEM	Compute inertial state of gravitational body at given time
5. CENTER	Convert state of bodies to barycentric coordinates
6. PECEQ	Compute transformation matrix from ecliptic to equatorial coordinates
7. EULMX	Compute rotation matrix
C. Propagation	
1. VMP	Executive routine for virtual mass trajectory propagation
2. ESTMT	Determine final position and magnitude of VM on current step
3. VECTOR	Compute the spacecraft final position on current step
4. VMASS	Determine VM data for current step
D. Input/Output	
1. TRAPAR	Compute and record navigation parameter data
2. INPUTZ	Convert input data into VMP compatible form
3. PRINT	Print output of VM trajectory
4. SPACE	Space paper for output purposes
5. NEWPGE	Record headings for each new page in VM printout
II. NOMNAL Subroutines	
A. Executive	
1. NOMNAL	Control nominal trajectory generation (Main program)
2. PRELIM	Perform preliminary work for NOMNAL
3. TRJTRY	Propagate virtual mass trajectory to next event
4. GIDANS	Control computation of trajectory correction
5. EXCUTE	Control execution of trajectory correction
B. Zero Iterate	
1. ZERIT	Control computation of zero iterate
2. HELIO	Compute heliocentric phase of interplanetary zero iterate
3. LAUNCH	Compute launch phase of interplanetary zero iterate
4. FLITE	Lambert time of flight equation solver
5. SERIE	Battin generalized function solver
6. LUNA	Control lunar zero iterate generation
7. LUNCON	Generate patched conic lunar trajectory
8. LUNTAR	Control patched conic targeting
9. MULCON	Generate lunar multi-conic trajectory

Table 4.2 STEAP II Subroutine Summaries (Cont'd)

10. MULTAR	Control multi-conic targeting
11. BATCON	Propagate conic trajectory
C. Targeting	
1. TARGET	Control n-body targeting
2. TAROPT	Set up target parameter arrays
3. TARMAX	Compute Newton-Raphson targeting matrix
4. DESENT	Compute steepest descent-conjugate gradient corrections
5. MATIN	Compute matrix inverse
D. Insertion	
1. INSERS	Control orbit insertion computations
2. COPINS	Compute coplanar orbit insertion
3. NONINS	Compute nonplanar orbit insertion
E. Pulsing Arc	
1. PREPUL	Perform preliminary work for multiple pulses
2. PULSEX	Execute pulsing arc
3. PERHEL	Propagate perturbed heliocentric conic
4. (BATCON)	Propagate conic trajectory
III. ERRAN and SIMUL Subroutines	
A. Executive	
1. ERRAN	Control error analysis program (Main program)
2. SIMUL	Control simulation program (Main program)
B. Dynamic Model	
1. NTM	Control generation of trajectory data for ERRAN
2. NTMS	Control generation of trajectory data for SIMUL
3. PSIM	Control computation of state transition matrix (STM)
4. NDTM	Compute unaugmented partitoin of STM by numerical differencing
5. PLND	Compute STM partition associated with ephemeris biases
6. MUND	Compute STM partition associated with gravitational constants
7. PCTM	Compute unaugmented partition of STM by patched conic technique
8. CONC2	Compute unaugmented partition of STM by virtual mass technique
9. CASCAD	Compute unaugmented partition of STM by cascaded Darby matrizants
C. Navigation	
1. NAVM	Propagate covariance matrices between measurements and between events
2. SCHED	Select next measurement time from measurement schedule
3. TRAKM	Compute observation matrices
4. TRAKS	Compute observation matrices and actual measurements
5. TARPRL	Compute target planet position partials
6. STAPRL	Compute station location position and velocity partials
7. MENO	Compute assumed measurement noise covariance matrix

Table 4.2 STEAP II Subroutine Summaries (Cont'd)

8. MENOS	Compute assumed and actual measurement noise covariance matrices
9. BIAS	Compute actual measurement bias
10. RNUM	Generate random numbers
11. DYNO	Compute dynamic noise covariance matrix
12. DYNOS	Compute dynamic noise covariance matrix and actual dynamic noise
13. GHA	Compute Greenwich hour angle
14. JACOBI	Compute eigenvalues and eigenvectors of a matrix
15. HYELS	Compute hyperellipsoids
16. EIGHY	Control computation of eigenvalues, eigenvectors, and hyperellipsoids
D. Event	
1. SETEVN	Perform computations common to most events in ERRAN
2. SETEVS	Perform computations common to most events in SIMUL
3. PRED	Perform prediction event in ERRAN
4. PRESIM	Perform prediction event in SIMUL
5. QUASI	Perform quasi-linear filtering event in SIMUL
6. GUIDM	Perform guidance event in ERRAN
7. GUISIM	Perform guidance event in SIMUL
8. GUID	Compute guidance and variation matrices in ERRAN
9. GUIS	Compute guidance and variation matrices in SIMUL
10. VARADA	Compute 3VBP variation matrix in ERRAN
11. VARSIM	Compute 3VBP variation matrix in SIMUL
12. PARTL	Compute partials of B•T, B•R wrt state
13. BIAIM	Perform biased aimpoint guidance
14. POICOM	Compute probability of impact
15. QCOMP	Compute execution error covariance matrix
16. NONLIN	Control execution of nonlinear guidance events
17. PULCOV	Propagate covariance matrix across a series of pulses
18. EXCUT	Control execution of pulsing arc in ERRAN
19. EXCUTS	Control execution of pulsing arc in SIMUL
E. Input/Output	
1. DATA	Perform preliminary computations and read data in ERRAN
2. DATA1	Continuation of DATA
3. DATAS	Perform preliminary computations and read data in SIMUL
4. DATAS1	Continuation of DATAS
5. CONVRT	Convert JPL injection conditions to Cartesian components
6. TRANS	Compute coordinate transformations
7. CORREL	Compute and print correlation matrix partitions and standard deviations
8. STMPR	Print STM partitions
9. SUB1	Compute position and velocity magnitudes
10. TITLE	Print titles
11. PRINT3	Print basic cycle data in ERRAN
12. PRINTS3	Print ERRAN summary
13. PRINT4	Print basic cycle data in SIMUL
14. PRINTS4	Print SIMUL summary

5. COMMON VARIABLE DEFINITIONS

THE BULK OF THE VARIABLES USED IN THE STEAP PROGRAMS ARE COMMON VARIABLES. THESE VARIABLES ARE DEFINED IN DETAIL IN THIS CHAPTER. THE FIRST SECTION LISTS THE COMMON BLOCKS IN ALPHABETICAL ORDER. THE PROGRAMS (NOMNAL, ERRAN, SIMUL) USING EACH COMMON BLOCK ARE NOTED. THE VARIABLES OF EACH COMMON BLOCK ARE DEFINED IN THE ORDER THAT THEY APPEAR IN THE COMMON BLOCK.

THE SECOND SECTION LISTS ALPHABETICALLY ALL VARIABLES APPEARING ANYWHERE IN COMMON. THE COMMON BLOCK TO WHICH THE VARIABLE BELONGS IS REFERENCED. THE DEFINITION OF THE VARIABLE IS THEN GIVEN.

THE THIRD SECTION SUPPLIES THE DEFINITIONS OF SEVERAL LARGE FREQUENTLY REFERENCED ARRAYS. THE ELEMENT APPEARING IN EACH COMPONENT OF EACH ARRAY IS NOTED.

5.1 COMMON VARIABLES BY BLOCKS

IN THIS SECTION COMMON BLOCKS APPEARING IN STEAP ARE LISTED IN ALPHABETICAL ORDER. VARIABLES WITHIN THESE BLOCKS ARE LISTED AND DEFINED IN THE ORDER THEY APPEAR IN THE PROGRAM.

/BAIM / MODE ERRAN, SIMUL

ATRANS(6)	CLOSEST APPROACH STATE
TMPR(3)	MOST RECENT TARGET STATE
TNOMC(7)	NOMINAL CLOSEST APPROACH TARGET STATE,INCL. TIME
TNOMB(3)	NOMINAL B-PLANE TARGET STATE
PHI2(3,3)	INVERSE OF VARIATION MATRIX PARTITION
VINF	HYPERBOLIC EXCESS VELOCITY
TINJ	INJECTION TIME
PROBI	ALLOWABLE PROBABILITY OF IMPACT
ADA(3,6)	VARIATION MATRIX
T3(10)	ARRAY OF GUIDANCE EVENT TIMES
IBAG	NOT USED
IPQ	NOT USED
IGUID(5,10)	ARRAY OF GUIDANCE EVENT CODES
II	GUIDANCE EVENT COUNTER

/BLK / MODE: NOMNAL, ERRAN, SIMUL

T	TRAJECTORY TIME IN DAYS
PMASS(11)	GRAVITATIONAL CONSTANTS OF PLANETS IN A.U.**3/DAY**2
CN(80)	CONSTANTS USED TO CALCULATE THE ORBITAL ELEMENTS OF THE FIRST FIVE PLANETS (SEE LARGE ARRAY DEFINITIONS IN SECTION 5.3)
ST(50)	CONSTANTS USED TO CALCULATE THE ORBITAL ELEMENTS OF THE LAST FOUR PLANETS (SEE LARGE ARRAY DEFINITIONS IN SECTION 5.3)
EMN(15)	THE CONSTANTS USED TO CALCULATE THE ORBITAL ELEMENTS OF THE MOON (SEE LARGE ARRAY DEFINITIONS IN SECTION 5.3)
SMJR(18)	CONSTANTS USED TO CALCULATE THE SEMI-MAJOR AXES OF THE PLANETS
RADIUS(11)	THE RADIUS OF A GIVEN PLANET IN A.U.
RMASS(11)	THE RELATIVE GRAVITATIONAL CONSTANT OF A STATED PLANET WITH RESPECT TO THE SUN
ELMNT(80)	CONTAINS THE ORBITAL ELEMENTS OF THE PLANETS (SEE LARGE ARRAY DEFINITIONS IN SECTION 5.3)
SPHERE(11)	THE SPHERES OF INFLUENCE OF THE PLANETS IN A.U.
XP(6)	THE POSITION AND VELOCITY OF A PLANET IN INERTIAL ECLIPTIC COORDINATES
NO(11)	AN ARRAY OF PLANET CODES BEING USED TO GENERATE THE VIRTUAL MASS TRAJECTORY

/CNTRIC/ MODE: NOMNAL, ERRAN, SIMUL

IBARY	REFERENCE COORDINATE SYSTEM CODE =0 HELIOCENTRIC COORDINATES =1 BARYCENTRIC COORDINATES
ICOORD	NON-FUNCTIONAL IN ERROR ANALYSIS MODE
INITIAL	NON-FUNCTIONAL IN ERROR ANALYSIS MODE

/COM / MODE: NOMNAL, ERRAN, SIMUL

V(16,7)	AN ARRAY WHICH STORES PERTINENT VECTORS USED IN THE CALCULATION OF THE VIRTUAL MASS TRAJECTORY (SEE LARGE ARRAY DEFNS IN SECT 5.3)
F(44,4)	CONTAINS THE POSITIONS AND VELOCITIES OF THE PLANETS AT A SPECIFIED TIME PLUS THE POSITIONS AND VELOCITIES OF THE SPACECRAFT RELATIVE TO THE PLANETS (SEE LARGE ARRAY DEFNS IN SECT 5.3)
PI	THE VALUE OF THE MATHEMATICAL CONSTANT PI
RAD	THE NUMBER OF DEGREES PER RADIAN
ITRAT	IN INTERNAL CODE USED TO DETERMINE HOW MANY ITERATIONS HAVE BEEN ACCOMPLISHED IN THE VIRTUAL MASS PROCEDURE
KOUNT	A CODE WHICH SPECIFIES WHETHER PRINT-OUT IS TO OCCUR AFTER THIS TIME INCREMENT
INCMNT	NUMBER OF INCREMENTS USED
INCPR	SPECIFIES AFTER HOW MANY TIME INCREMENTS PRINT-OUT IS TO OCCUR
INC	DETERMINE WHETHER THE ABOVE OPTION IS TO BE USED
IPR	A CODE WHICH DETERMINES IF PRINT-OUT IS TO OCCUR AFTER A SPECIFIED NUMBER OF DAYS

NBODYI	NUMBER OF BODIES CONSIDERED IN VIRTUAL MASS TRAJECTORY
NBODY	BASED ON ABOVE VALUE--EQUAL TO 4*NBODYI-3
IPRT(4)	SPECIFIES PRINT OPTIONS (IN STEAP TRAJECTORY THIS OPTION IS OMITTED. WHEN PRINT-OUT OCCURS ALL SECTIONS ARE AUTOMATICALLY PRINTED)
KL	PROBLEM NUMBER (NOMNAL ONLY)
IPG	PAGE NUMBER (NOMNAL ONLY)
LINCT	LINE COUNT (NOMNAL ONLY)
LINPGE	LINES PER PAGE (NOMNAL ONLY)

/CONST / MODE: ERRAN, SIMUL

OMEGA	ROTATION RATE OF EARTH
EPS	OBLIQUITY OF EARTH
SAL(3)	ALTITUDES OF STATIONS
SLAT(3)	LATITUDES OF STATIONS
SLON(3)	LONGITUDES OF STATIONS
DNCN(3)	CONSTANTS FROM WHICH DYNAMIC NOISE IS COMPUTED
MNCN(12)	MEASUREMENT NOISE CONSTANTS
NST	NUMBER OF STATIONS TO BE USED (MAXIMUM 3)

/CONST2/ MODE: ERRAN, SIMUL	
UST(3)	DIRECTION COSINE ARRAYS OF THREE REFERENCE STARS
VST(3)	DIRECTION COSINE ARRAYS OF THREE REFERENCE STARS
WST(3)	DIRECTION COSINE ARRAYS OF THREE REFERENCE STARS
FOP	OFF-DIAGONAL ANNIHILATION VALUE FOR POSITION EIGENVALUES
FOV	OFF-DIAGONAL ANNIHILATION VALUE FOR VELOCITY EIGENVALUES

/CONST3/ MODE: ERRAN, SIMUL	
DELAXS	TARGET PLANET SEMI-MAJOR AXIS FACTOR USED IN NUMERICAL DIFFERENCING
DELECC	TARGET PLANET ECCENTRICITY FACTOR USED IN NUMERICAL DIFFERENCING
DELICL	TARGET PLANET INCLINATION FACTOR USED IN NUMERICAL DIFFERENCING
DELNOD	TARGET PLANET LONGITUDE OF THE ASCENDING NODE FACTOR USED IN NUMERICAL DIFFERENCING
DELW	TARGET PLANET ARGUMENT OF PERIAPSIS FACTOR USED IN NUMERICAL DIFFERENCING
DELMA	TARGET PLANET MEAN ANOMALY FACTOR USED IN NUMERICAL DIFFERENCING
DELMUS	SUN GRAVITATIONAL CONSTANT FACTOR USED IN NUMERICAL DIFFERENCING
DELMUP	TARGET PLANET GRAVITATIONAL CONSTANT FACTOR USED IN NUMERICAL DIFFERENCING

/OPNUM / MODE: NOMNAL

ZERO	THE NUMBER ZERO (0) TO NINE SIGNIFICANT FIGURES
ONE	THE NUMBER ONE (1) TO NINE SIGNIFICANT FIGURES
TWO	THE NUMBER TWO (2) TO NINE SIGNIFICANT FIGURES
THREE	THE NUMBER THREE (3) TO NINE SIGNIFICANT FIGURES
FOUR	THE NUMBER FOUR (4) TO NINE SIGNIFICANT FIGURES
FIVE	THE NUMBER FIVE (5) TO NINE SIGNIFICANT FIGURES
EIGHT	THE NUMBER EIGHT (8) TO NINE SIGNIFICANT FIGURES
TEN	THE NUMBER TEN (10) TO NINE SIGNIFICANT FIGURES
NINETY	THE NUMBER NINETY (90) TO NINE SIGNIFICANT FIGURES
HALF	THE NUMBER ONE-HALF (1/2) TO NINE SIGNIFICANT FIGURES

/DPNUM / MODE: ERRAN, SIMUL

ZERO	THE NUMBER ZERO (0) TO NINE SIGNIFICANT FIGURES
ONE	THE NUMBER ONE (1) TO NINE SIGNIFICANT FIGURES
TWO	THE NUMBER TWO (2) TO NINE SIGNIFICANT FIGURES
HALF	THE NUMBER ONE-HALF (1/2) TO NINE SIGNIFICANT FIGURES
THREE	THE NUMBER THREE (3) TO NINE SIGNIFICANT FIGURES
EM1	THE NUMBER 1.E-1 TO NINE SIGNIFICANT FIGURES
EM2	THE NUMBER 1.E-2 TO NINE SIGNIFICANT FIGURES
EM3	THE NUMBER 1.E-3 TO NINE SIGNIFICANT FIGURES
EM4	THE NUMBER 1.E-4 TO NINE SIGNIFICANT FIGURES
EM5	THE NUMBER 1.E-5 TO NINE SIGNIFICANT FIGURES
EM6	THE NUMBER 1.E-6 TO NINE SIGNIFICANT FIGURES
EM7	THE NUMBER 1.E-7 TO NINE SIGNIFICANT FIGURES
EM8	THE NUMBER 1.E-8 TO NINE SIGNIFICANT FIGURES
EM9	THE NUMBER 1.E-9 TO NINE SIGNIFICANT FIGURES
EM50	THE NUMBER 1.E-50 TO NINE SIGNIFICANT FIGURES
TWOPI	THE MATHEMATICAL CONSTANT 2.*PI
EM13	THE NUMBER 1.E-13 TO NINE SIGNIFICANT FIGURES

/EVENT / MODE: ERRAN, SIMUL

TEV(50)	TIMES OF EVENTS
TPT2(20)	PREDICTION TIMES
SIGRES	VARIANCE OF RESOLUTION ERROR
SIGPRO	VARIANCE OF PROPORTIONALITY ERROR
SIGALP	VARIANCE OF ERROR IN POINTING ANGLE 1
SIGBET	VARIANCE OF ERROR IN POINTING ANGLE 2
HP7	ORBIT INSERTION VARIABLES. NON-FUNCTIONAL IN EXISTING PROGRAM
P7	ORBIT INSERTION VARIABLES. NON-FUNCTIONAL IN EXISTING PROGRAM
TAU7	ORBIT INSERTION VARIABLES. NON-FUNCTIONAL IN EXISTING PROGRAM
AINC7	ORBIT INSERTION VARIABLES. NON-FUNCTIONAL IN EXISTING PROGRAM
ANODE7	ORBIT INSERTION VARIABLES. NON-FUNCTIONAL IN EXISTING PROGRAM
PERP7	ORBIT INSERTION VARIABLES. NON-FUNCTIONAL IN EXISTING PROGRAM
ECC7	ORBIT INSERTION VARIABLES. NON-FUNCTIONAL IN EXISTING PROGRAM
DV8(3)	ORBIT INSERTION VARIABLE. NON-FUNCTIONAL IN EXISTING PROGRAM
NEV	NUMBER OF EVENTS
IEVNT(50)	CODES OF EVENTS
IHYP1	HYPERELLIPSOID CODE USED TO DETERMINE IF K=1, K=3, OR BOTH

IEIG	CODE USED TO DECIDE IF BOTH POSITION AND VELOCITY EIGENVECTORS ARE REQUESTED
ICDT3(20)	CODES WHICH DETERMINE WHICH GUIDANCE POLICIES ARE BEING USED
NPE	NUMBER OF PREDICTION EVENTS HAVING OCCURRED
NGE	NUMBER OF GUIDANCE EVENTS HAVING OCCURRED
IPOL	CODE WHICH DETERMINES IF FIXED-TIME-OF-ARRIVAL GUIDANCE EVENT HAS OCCURED
IIPOL	CODE WHICH DETERMINES IF EITHER TWO-VARIABLE OR THREE-VARIABLE B-PLANE GUIDANCE POLICY HAS OCCURRED
ICDQ3(20)	ARRAY OF CODES WHICH DETERMINE WHICH EXECUTION POLICIES ARE TO BE USED IN GUIDANCE EVENTS
NEV1	TOTAL NUMBER OF EIGENVECTOR EVENTS
NEV2	TOTAL NUMBER OF PREDICTION EVENTS
NEV3	TOTAL NUMBER OF GUIDANCE EVENTS
NEV4	TOTAL NUMBER OF -COMCON- EVENTS
NQE	QUASI-LINEAR FILTERING EVENTS HAVING OCCURRED
NEV5	TOTAL NUMBER OF QUASI-LINEAR FILTERING EVENTS
NEV6	TOTAL NUMBER OF ADAPTIVE FILTERING EVENTS. NON-FUNCTIONAL IN EXISTING PROGRAM
NAE	ADAPTIVE FILTERING EVENTS HAVING OCCURRED. NON-FUNCTIONAL IN EXISTING PROGRAM
NAF6(20)	ARRAY OF ADAPTIVE FILTERING EVENT CODES. NON-FUNCTIONAL IN EXISTING PROGRAM
NEV7	ORBIT INSERTION VARIABLES. NON-FUNCTIONAL IN EXISTING PROGRAM

IOPT7	ORBIT INSERTION VARIABLE. NON-FUNCTIONAL IN EXISTING PROGRAM
NEV8	ORBIT INSERTION VARIABLE. NON-FUNCTIONAL IN EXISTING PROGRAM
NEV9	ORBIT INSERTION VARIABLES. NON-FUNC IONAL IN EXISTING PROGRAM
NEV10	ORBIT INSERTION VARIABLES. NON-FUNCTIONAL IN EXISTING PROGRAM
NEV11	ORBIT INSERTION VARIADLES. NON-FUNCTIONAL IN EXISTING PROGRAM

/EXE / MODE ERRAN, SIMUL

XXIN(6)	STATE VECTOR TRANSFERRED TO EXCUT OR EXCUTS
DIPX	JULIAN DATE TRANSFERRED TO EXCUT OR EXCUTS
DELPX(3)	VELOCITY CORRETION TO BE MODELED AS AN IMPULSE SERIES
QK(6,6)	EFFECTIVE EXECUTION COVARIANCE MATRIX
DUMMYQ(6)	ARRAY OF EXECUTION ERROR VARIANCES
INPX	IMPUSLE SERIES CODE

/GUI / MODE: ERRAN, SIMUL	
PG(6,6)	POSITION/VELOCITY CONTROL COVARIANCE
CXXSG(6,24)	CONTROL CORRELATION BETWEEN POSITION/VELOCITY STATE AND SOLVE-FOR PARAMETERS
CXUG(6,8)	CONTROL CORRELATION BETWEEN POSITION/VELOCITY STATE AND DYNAMIC CONSIDER PARAMETERS
CXVG(6,15)	CONTROL CORRELATION BETWEEN POSITION/VELOCITY STATE AND MEASUREMENT CONSIDER PARAMETERS
PSG(24,24)	SOLVE-FOR PARAMETER CONTROL COVARIANCE
CXSUG(24,8)	CONTROL CORRELATION BETWEEN SOLVE-FOR PARAMETERS AND DYNAMIC CONSIDER PARAMETERS
CXSVG(24,15)	CONTROL CORRELATION BETWEEN SOLVE-FOR PARAMETERS AND MEASUREMENT CONSIDER PARAMETERS
XG(6)	POSITION/VELOCITY STATE AT MOST RECENT GUIDANCE EVENT
TG	TRAJECTORY TIME AT MOST RECENT GUIDANCE EVENT
EM(2,6)	VARIATION MATRIX RELATING POSITION/VELOCITY DEVIATIONS TO B.T AND B.R DEVIATIONS

/IMPTAR/ MODE NOMNAL, ERRAN, SIMUL	
ANG	TARGET INCLINATION CONVERTED FROM INPUT FORMAT TO VALUE BETWEEN 0 AND 180 DEGREES AND SATISFYING APPROACH ASYMPTOTE CONSTRAINT

/LUNART/ MODE NOMNAL

DTAR(3)	TARGET VALUES OF SMA, B.T, AND B.R IN LUNAR TARGETING
PCON(3)	PERTURBATIONS IN CONTROLS (ALPHA,DELTA,THETA)
TTOL(3)	ALLOWABLE TOLERANCES IN SMA, B.T, B.R
BCON(3)	MAXIMUM STEP SIZES OF CONTROLS
RI(6)	GEOCENTRIC STATE OF S/C AT LUNAR SOI
RMQ(6)	GEOCENTRIC STATE OF CENTER OF MOON AT TSI IN EQUATORIAL COORDINATES
RSI(6)	SELENOCENTRIC STATE OF S/C AT LUNAR SOI
RME(6)	GEOCENTRIC STATE OF CENTER OF MOON IN ECLIPTIC COORDINATES AT TSI
DECLIN	DECLINATION OF APPROACH ASYMPTOTE WITH RESPECT TO LUNAR EQUATOR
OTAR(3)	DESIRED VALUES OF SMA, RCA, AND INC
TCA	J.D. OF TIME AT LUNAR CLOSEST APPROACH (DESIRED)
RCA	RADIUS OF CLOSEST APPROACH TO MOON (DESIRED)
SMA	SEMI-MAJOR AXIS OF LUNAR HYPERBOLA (DESIRED)
CAI	DESIRED CLOSEST APPROACH EQUATORIAL INCLINATION
RPE	RADIUS OF EARTH PARKING ORBIT
TSI	PROJECTED J.D. AT SOI INTERSECTION
EMU	GRAVITATIONAL CONSTANT OF EARTH (KM3/SEC2)

TSPH	RADIUS OF LUNAR SOI (KM)
EQLQ(3,3)	TRANSFORMATION MATRIX FROM EARTH-EQUATORIAL TO LUNAR EQUATORIAL COORDINATES
ITAG	FLAG SPECIFYING STAGE OF TARGETING =1 IN SMA TARGETING =0 IN SMA, INC, RCA TARGETING

/MEAS / MODE: ERRAN, SIMUL

TMN(1000)	TIMES OF MEASUREMENTS
MCODE(1000)	ARRAY OF MEASUREMENT CODES
NMN	TOTAL NUMBER OF 'EASUREMENTS
MCNTR	NUMBER OF MEASUR ENTS HAVING OCCURRED

/MISC / MODE: ERRAN, SIMUL

ACC	ACCURACY FIGURE USED IN VIRTUAL MASS PROGRAM
FACP	POSITION FACTOR USED IN NUMERICAL DIFFERENCING
FACV	VELOCITY FACTOR USED IN NUMERICAL DIFFERENCING
BIA(12)	MEASUREMENT BIASES
IDNF	DYNAMIC NOISE FLAG
ICOOR	STATE VECTOR CODE WHICH DETERMINES IN WHICH COORDINATE SYSTEM THE VECTOR IS READ IN
ITR	MODE FLAG
IMNF	MEASUREMENT NOISE FLAG
ISP2	SPHERE OF INFLUENCE FLAG

/NAME / MODE: ERRAN, SIMUL

EVNM(11)	EVENT NAME
MNNAME(12,3)	MEASUREMENT NAME
CMPNM(30)	COMPONENT NAME

/OVER / MODE: SIMUL

RF(6)	FINAL TARGETED NOMINAL STATE VECTOR
RF1(6)	FINAL MOST RECENT NOMINAL STATE VECTOR

/OVER1 / MODE: SIMUL

RI(6)	INITIAL TARGETED NOMINAL STATE VECTOR
TEVN	TIME OF CURRENT EVENT
RI1(6)	INITIAL MOST RECENT NOMINAL STATE VECTOR
ICODE	EVENT CODE
NAFC	NON-FUNCTIONAL ADAPTIVE FILTER CODE
NR	NUMBER OF ROWS IN THE OBSERVATICN MATRIX

/OVERL / MODE ERRAN, SIMUL

DTIME	TIME INTERVAL BETWEEN ORBITAL INSERTION DECISION AND EXECUTION

/OVERR / MODE ERRAN

RI(6)	STATE VECTOR AT EVENT TIME
TEVN	EVENT TIME

/OVERX / MODE ERRAN, SIMUL

IX	NONLINEAR GUIDANCE CODE
JX	GUIDANCE EVENT COUNTER
XIN(6)	STATE VECTOR TRANSFERRED TO NONLIN

/OVERZ / MODE ERRAN, SIMUL

RF(6)	FINAL TARGETED STATE VECTOR
RF1(6)	FINAL MOST RECENT NOMINAL STATE VECTOR
IGP	MIDCOURSE GUIDANCE POLICY CODE
GA(3,6)	GUIDANCE MATRIX

/PBLK / MODE ERRAN, SIMUL

A(2,3)	FTA IMPACT PLANE TRANSFORMATION MATRIX
XMUS(2)	NOMINAL IMPACT PLANE TARGET STATE
EXEC(3,3)	EXECUTION ERROR COVARIANCE MATRIX
CR	CAPTURE RADIUS OF TARGET PLANET
POI	PROBABILITY OF IMPACT
XLAM(2,2)	PROJECTION OF TARGET CONDITION COVARIANCE MATRIX INTO THE IMPACT PLANE
XLAMI(2,2)	INVERSE OF XLAM(2,2)
DVRB(3)	VELOCITY CORRECTION REQUIRED TO REMOVE AIMPOINT BIAS
DVUP(3)	UPDATE VELOCITY CORRECTION
PSTAR	NOMINAL PROBABILITY DENSITY FUNCTION EVALUATED AT TARGET PLANET CENTER
DVN(3)	COMMANDED VELOCITY CORRECTION TRANSFERRED TO BIAIM
DELV(3,10)	ARRAY OF EXTERNALLY-SUPPLIED VELOCITY CHANGES
IIGP	MIDCOURSE GUIDCANCE POLICY CODE
IBIAS	BIASED AIMPOINT GUIDANCE EVENT FLAG = 0 AIMPOINT NOT BIASED = 1 AIMPOINT BIASED
IDENS	PROBABILITY DENSITY FUNCTION CODE. NON-FUNCTIONAL

/PRT	/ MODE	NOMNAL

MONTH(12)	NAMES OF MONTHS
PLANET(11)	NAMES OF GRAVITATIONAL BODIES

/PRT	/ MODE:	ERRAN, SIMUL

PLANET(11)	NAMES OF PLANETS

/PULS / MODE NOMNAL

PULMAG	THRUST MAGNITUDE OF PULSING ENGINE
PULMAS	NOMINAL MASS OF SPACECRAFT DURING PULSING ARC
DUR	DURATION OF SINGLE PULSE
DTI	TIME INTERVAL (DAYS) BETWEEN SUCCESSIVE PULSES
DVI(3)	VELOCITY INCREMENT ADDED ON TYPICAL PULSE
DVF(3)	VELOCITY INCREMENT ADDED ON FINAL PULSE
PULT	TOTAL TIME INTERVAL OF PULSING ARC
RK(2,3)	POSITION VECTORS OF LAUNCH AND TARGET PLANETS AT IMPULSIVE TIME (MIDPOINT OF PULSING ARC)
VK(2,3)	VELOCITY VECTORS OF LAUNCH AND TARGET PLANETS AT IMPULSIVE TIME (MIDPOINT OF PULSING ARC)
FS(2,5)	F-SERIES COEFFICIENTS OF LAUNCH AND TARGET BODIES
GS(2,4)	G-SERIES COEFFICIENTS OF LAUNCH AND TARGET BODIES
GG(3)	GRAVITATIONAL CONSTANTS OF SUN, LAUNCH, AND TARGET BODIES
NPUL	NUMBER OF PULSES IN PULSING ARC

/SAVVAL/ MODE ERRAN, SIMUL

XBDT	ORIGINAL VALUE OF B.T IN NONLINEAR GUIDANCE
XBDR	ORIGINAL VALUE OF B.R IN NONLINEAR GUIDANCE
XDSI	ORIGINAL VALUE OF TSI IN NONLINEAR GUIDANCE
XRSI(3)	ORIGINAL VALUE OF RSI IN NONLINEAR GUIDANCE
XVSI(3)	ORIGINAL VALUE OF VSI IN NONLINEAR GUIDANCE
XRC(6)	ORIGINAL VALUE OF RC IN NONLINEAR GUIDANCE
XDC	ORIGINAL VALUE OF DC IN NONLINEAR GUIDANCE

/SIMCNT/ MODE: SIMUL

DMUSB	BIAS IN GRAVITATIONAL CONSTANT OF SUN
DMUPB	BIAS IN GRAVITATIONAL CONSTANT OF TARGET PLANET
DAB	BIAS IN SEMI-MAJOR AXIS OF TARGET PLANET
DEB	BIAS IN ECCENTRICITY OF TARGET PLANET
DIB	BIAS IN INCLINATION OF TARGET PLANET
DNOB	BIAS IN LONGITUDE OF ASCENDING NODE
DWB	BIAS IN ARGUMENT OF PERIAPSIS
DMAB	BIAS IN MEAN ANOMALY
TTIM1	FIRST TIME USED FOR UNMODELLED ACCELERATION
TTIM2	SECOND TIME USED FOR UNMODELLED ACCELERATION

UNMAC(3,3)	UNMODELLED ACCELERATION
SLB(9)	BIASES IN STATION LOCATION CONSTANTS
AVARM(12)	VARIANCE OF ACTUAL MEASUREMENT NOISE
ARES(20)	ACTUAL RESOLUTION ERROR
APRO(20)	ACTUAL PROPORTIONALITY ERROR
AALP(20)	ACTUAL ERROR IN POINTING ANGLE 1
ABET(20)	ACTUAL ERROR IN POINTING ANGLE 2
IAMNF	ACTUAL MEASUREMENT NOISE FLAG

/SIM1 / MODE: SIMUL

XI1(6)	INITIAL STATE VECTOR OF MOST RECENT NOMINAL TRAJECTORY
XF1(6)	FINAL STATE VECTOR OF MOST RECENT NOMINAL TRAJECTORY
ADEVX(6)	ACTUAL DEVIATION IN THE STATE VECTOR
ADEVXS(24)	ACTUAL DEVIATION IN SOLVE-FOR PARAMETERS
EDEVX(6)	ESTIMATED DEVIATION IN THE STATE VECTOR
EDEVXS(24)	ESTIMATED DEVIATION IN SOLVE-FOR PARAMETERS
W(6)	ACTUAL DYNAMIC NOISE
ZI(6)	INITIAL ACTUAL STATE VECTOR
ZF(6)	FINAL ACTUAL STATE VECTOR AFTER ADDING EFFECT OF UNMODELED ACCELERATION
ANOIS(4)	ACTUAL WHITE NOISE
RES(4)	RESIDUAL
EY(4)	ESTIMATED MEASUREMENT
AY(4)	ACTUAL MEASUREMENT
AR(4,4)	ACTUAL MEASUREMENT NOISE
ADEVXB(6)	ACTUAL DEVIATION IN STATE VECTOR AT BEGINNING OF TRAJECTORY
ADEVSB(24)	ACTUAL DEVIATION IN SOLVE-FOR PARAMETERS AT BEGINNING OF TRAJECTORY
AYMEY(4)	ACTUAL MEASUREMENT MINUS ESTIMATED MEASUREMENT
EDEVXM(6)	ESTIMATED DEVIATION IN THE STATE VECTOR (FOR ADAPTIVE FILTERING)
EDEVSM(24)	ESTIMATED DEVIATION IN SOLVE-FOR PARAMETERS (FOR ADAPTIVE FILTERING)

/SIM2 / MODE: SIMUL

NB1(11)	ARRAY OF PLANET CODES IN ACTUAL TRAJECTORY
ACC1	ACCURACY USED IN ACTUAL TRAJECTORY
NBOD1	NUMBER OF BODIES IN ACTUAL TRAJECTORY

/STM / MODE: ERRAN, SIMUL

P(6,6)	POSITION/VELOCITY COVARIANCE
CXXS(6,24)	CORRELATION BETWEEN POSITION/VELOCITY STATE AND SOLVE-FOR PARAMETERS
CXU(6,8)	CORRELATION BETWEEN POSITION/VELOCITY STATE AND DYNAMIC CONSIDER PARAMETERS
CXV(6,15)	CORRELATION BETWEEN POSITION/VELOCITY STATE AND MEASUREMENT CONSIDER PARAMETERS
PS(24,24)	SOLVE-FOR PARAMETER COVARIANCE
CXSU(24,8)	CORRELATION BETWEEN SOLVE-FOR PARAMETERS AND DYNAMIC CONSIDER PARAMETERS
CXSV(24,15)	CORRELATION BETWEEN SOLVE-FOR PARAMETERS AND MEASUREMENT CONSIDER PARAMETERS
UO(8,8)	DYNAMIC CONSIDER PARAMETER COVARIANCE MATRIX

V0(15,15)	MEASUREMENT CONSIDER PARAMETER COVARIANCE MATRIX

NOTE IF THE ENTIRE COVARIANCE MATRIX WERE ASSEMBLED FROM THE GIVEN PARTITIONS THE RESULTANT MATRIX WOULD BE P(53,53) WITH THE SYMMETRIC STRUCTURE-

	P(6,6)	CXXS(6,24),	CXU(6,8)	CXV(6,15)
P(53,53)=		PS(24,24)	CXSU(24,8)	CXSV(24,15)
			U0(8,8)	CUV(8,15)
				V0(15,15)

PHI(6,6)	POSITION/VELOCITY STATE TRANSITION MATRIX
TXXS(6,24)	STATE TRANSITION MATRIX PARTITION ASSOCIATED WITH SOLVE-FOR PARAMETERS
TXU(6,8)	STATE TRANSITION MATRIX PARTITION ASSOCIATED WITH DYNAMIC CONSIDER PARAMETERS
Q(6,6)	DYNAMIC NOISE COVARIANCE MATRIX
R(4,4)	MEASUREMENT NOISE COVARIANCE MATRIX
AK(6,4)	KALMAN GAIN CONSTANT FOR POSITION/VELOCITY STATE
S(24,4)	KALMAN GAIN CONSTANT FOR SOLVE-FOR PARAMETERS
H(4,6)	OBSERVATION MATRIX RELATING OBSERVABLES TO POSITION/VELOCITY STATE
AM(4,24)	OBSERVATION MATRIX RELATING OBSERVABLES TO SOLVE-FOR PARAMETER STATE
G(4,8)	OBSERVATION MATRIX RELATING OBSERVABLES TO DYNAMIC CONSIDER PARAMETER STATE

AL(4,15)	OBSERVATION MATRIX RELATING OBSERVABLES TO MEASUREMENT CONSIDER PARAMETER STATE
HPHR(4,4)	NON-FUNCTIONAL IN PRESENT ERROR ANALYSIS PROGRAM
PB(6,6)	POSITION/VELOCITY COVARIANCE AT INITIAL TIME
CXXSB(6,24)	CORRELATION BETWEEN POSITION/VELOCITY STATE AND SOLVE-FOR PARAMETERS AT INITIAL TIME
CXUB(6,8)	CORRELATION BETWEEN POSITION/VELOCITY STATE AND DYNAMIC CONSIDER PARAMETERS AT INITIAL TIME
CXVB(6,15)	CORRELATION BETWEEN POSITION/VELOCITY STATE AND MEASUREMENT CONSIDER PARMETERS AT INITIAL TIME
PSB(24,24)	SOLVE-FOR PARAMETER COVARIANCE AT INITIAL TIME
CXSUB(24,8)	CORRELATION BETWEEN SOLVE-FOR PARAMETERS AND DYNAMIC CONSIDER PARAMETERS AT INITIAL TIME
CXSVB(24,15)	CORRELATION BETWEEN SOLVE-FOR PARAMETERS AND MEASUREMENT CONSIDER PARAMETERS AT INITIAL TIME
PP(6,6)	POSITION/VELOCITY COVARIANCE MATRIX PRIOR TO PROCESSING A MEASUREMENT
CXXSP(6,24)	CORRELATION BETWEEN POSITION/VELOCITY STATE AND SOLVE-FOR PARAMETERS PRIOR TO PROCESSING A MEASUREMENT
CXUP(6,8)	CORRELATION BETWEEN POSITION/VELOCITY STATE AND DYNAMIC CONSIDER PARAMETERS PRIOR TO PROCESSING A MEASUREMENT
CXVP(6,15)	CORRELATION BETWEEN POSITION/VELOCITY STATE AND MEASUREMENT CONSIDER PARAMETERS PRIOR TO PROCESSING A MEASUREMENT
PSP(24,24)	SOLVE-FOR PARMETER COVARIANCE MATRIX PRIOR TO PROCESSING A MEASUREMENT

CXSUP(24,8)	CORRELATION BETWEEN SOLVE-FOR PARAMETERS AND DYNAMIC CONSIDER PARAMETERS PRIOR TO PROCESSING A MEASUREMENT
CXSVP(24,15)	CORRELATION BETWEEN SOLVE-FOR PARAMETERS AND MEASUREMENT CONSIDER PARAMETERS PRIOR TO PROCESSING A MEASUREMENT

/STVEC / MODE: ERRAN, SIMUL

XI(6)	INITIAL VEHICLE STATE VECTOR OF ORIGINAL NOMINAL
XF(6)	FINAL VEHICLE STATE VECTOR OF ORIGINAL NOMINAL
XB(6)	BEGINNING ORIGINAL NOMINAL VEHICLE STATE VECTOR
NDIM1	DIMENSION OF SOLVE-FOR PARAMETER STATE
NDIM2	DIMENSION OF DYNAMIC CONSIDER STATE
NDIM3	DIMENSION OF MEASUREMENT CONSIDER PARAMETER STATE
IAUGIN(24)	INPUT AUGMENTATION VECTOR OF ONE*S AND ZERO*S
IAUG(24)	AUGMENTATION VECTOR
IAUGDC(8)	DYNAMIC CONSIDER AUGMENTATION VECTOR
IAUGMC(15)	MEASUREMENT CONSIDER AUGMENTATION VECTOR

/TAREAL/ MODE: NOMNAL, ERRAN, SIMUL	
AC(5,10)	ACCURACY LEVELS (UP TO 5) USED IN EACH GUIDANCE EVENT
PHI(3,3)	TARGETING MATRIX
TIMG(10)	TIMES OF EACH GUIDANCE EVENT REFERENCED TO EPOCH--INITIAL TIME, SOI TIME, OR CA TIME
TAR(6,10)	DESIRED VALUES OF TARGET PARAMETERS (UP TO 6 AVAILABLE) FOR EACH GUIDANCE EVENT
DAUX(3)	DESIRED AUXILIARY PARAMETER VALUES OF ITERATE
AAUX(3)	ACTUAL AUXILIARY VALUES OF ITERATE
DTAR(3)	DESIRED TARGET VALUES OF ITERATE
ATAR(3)	ACTUAL TARGET VALUES OF ITERATE
TOL(6,10)	ALLOWABLE TOLERANCES OF TARGET PARAMETERS FOR EACH GUIDANCE EVENT
TOLR(6)	NOT USED IN CURRENT TARGET VERSION
CTOL(6)	TOLERANCES FOR CURRENT EVENT
FAC(3)	SCALING FACTORS USED IN BAD STEP CHECK
THPR	DAYS BETWEEN PRINTOUTS OF NOMINAL TRAJECTORY
PERV(10)	PERTURBATION SIZE FOR VELOCITY COMPONENTS IN CONSTRUCTING SENSITIVITY MATRICES IN TARGETING EVENTS
DINTG(10)	NOT USED IN CURRENT TARGET VERSION
DT(10)	JULIAN DATES OF TARGET TIMES
DELV(3,10)	EXTERNALLY SUPPLIED VELOCITY CORRECTION OR VELOCITY INCREMENT COMPUTED BY INSERTION DECISION

TRTM	TRAJECTORY TIME (DAYS) REF. TO INJECTION
RIN(6)	CURRENT STATE VECTOR AT I-TH EVENT
TIN	JULIAN DATE AT INJECTION
D1	JULIAN DATE ASSOCIATED WITH RIN ARRAY
DG(10)	JULIAN DATES OF EVENT TIMES
DELTAT	NUMBER OF DAYS INTEGRATION IS TO CONTINUE IF NO OTHER STOPPING CONDITION OCCURS
TMU	GRAVITATIONAL CONSTANT OF TARGET PLANET
RRF(3)	SPACECRAFT POSITION AT END OF INTEGRATION
DELTAV(3)	CORRECTIONS TO BE ADDED TO VELOCITY COMPONENTS FOR NEXT ITERATION
DVMAX(10)	MAXIMUM ALLOWABLE CHANGE IN ANY VELOCITY COMPONENT FOR EACH EVENT
ACKT	TRAJECTORY INTEGRATION ACCURACY
EQECP(3,3)	TRANSFORMATION FROM ECLIPTIC TO EQUATORIAL SYSTEM FOR TARGET PLANE
TIMS	INTERNAL CLOCK TIME AT START OF COMPUTER RUN
SPHFAC(10)	REDUCTION FACTORS FOR TARGET PLANET SPHERE OF INFLUENCE FOR EACH EVENT

/TARINT/ MODE: NOMNAL, ERRAN, SIMUL

NOGYD	TOTAL NUMBER OF GUIDANCE EVENTS
KTIM(10)	EPOCH TO WHICH GUIDANCE EVENT TIMES ARE REFERENCED =0 EVENT NOT PROCESSED =1 INITIAL TIME =2 SOI TIME =3 CA TIME =4 CALENDAR DATE
KTYP(10)	TYPE OF GUIDANCE EVENT FOR EACH EVENT =-1 TERMINATION EVENT =1 TARGETING EVENT =2 RETARGETING EVENT =3 ORBIT INSERTION EVENT
KMXQ(10)	COMPUTE/EXECUTE MODES FOR EACH GUIDANCE EVENT =1 COMPUTE VELOCITY CORRECTION ONLY =2 EXECUTE VELOCITY CORRECTION ONLY =3 COMPUTE AND IMMEDIATELY EXECUTE CORRECTION =4 COMPUTE BUT EXECUTE CORRECTION LATER
MDL(10)	EXECUTION MODELS FOR EACH GUIDANCE EVENT =1 IMPULSIVE =2 PULSING ARC
NPAR(10)	NUMBER OF TARGET PARAMETERS IN EACH TARGETING EVENT
KTAR(6,10)	CODES OF TARGET PARAMETERS (UP TO 6) FOR EACH TARGETING EVENT OR ORBIT INSERTION OPTION FOR EACH INSERTION EVENT
KEYTAR(3)	KEY DEFINING DESIRED TARGET PARAMETERS FOR CURRENT EVENT
MAT(10)	TARGETING MATRIX COMPUTATION CODE FOR EACH TARGETING EVENT =1 COMPUTE TARGETING MATRIX ONLY AT FIRST LEVEL =2 COMPUTE TARGETING MATRIX AT EACH STEP

IBADS(10)	BAD STEP FLAGS FOR EACH TARGETING EVENT =1 NEVER USE BAD STEP CHECK =2 USE BAD STEP CHECK AT FINAL LEVEL ONLY =3 USE BAD STEP CHECK AT ALL LEVELS
NOIT(10)	THE NUMBER OF TOTAL ITERATIONS ALLOWED AT THE FIRST AND LAST LEVELS OF TARGETING EVENTS FOR EACH GUIDANCE EVENT
MAXB(10)	THE NUMBER OF BAD STEPS ALLOWED DURING ANY TARGETING EVENT
LEVELS	NUMBER OF ACCURACY LEVELS FOR CURRENT EVENT
LEV	CURRENT LEVEL IN CURRENT TARGETING EVENT
NITS	ALLOWABLE NUMBER OF ITERATIONS FOR CURRENT EVENT
MAXBAD	MAXIMUM NUMBER OF BAD ITERATIONS FOR CURRENT EVENT
IBAST	BAD STEP CHECK INDICATOR FOR CURRENT EVENT
MATX	MATRIX COMPUTATION CODE FOR CURRENT TARGETING EVENT (SEE DEFN OF MAT)
ISTART	STAGE OF INITIAL TARGETING =0 NO TARGETING STARTED =1 FIRST PHASE STARTED AND HAVE TARGETING MATRIX =2 SECOND PHASE STARTED AND HAVE MATRIX
IPHASE	PHASE COUNTER FOR CURRENT TARGETING EVENT
NOPHAS	NUMBER OF TARGETING PHASES FOR CURRENT EVENT
ITARM	FLAG TO CONTROL CONSTRUCTION OF TARGETING MATRIX =0 DO NOT COMPUTE TARGETING MATRIX =1 COMPUTE TARGETING MATRIX ON CURRENT ITERATION
IBAD	BAD STEP FLAG FOR CURRENT ACCURACY LEVEL =1 DO NOT CHECK FOR BAD STEP =2 CHECK FOR BAD STEP

ISTOP	STOPPING CONDITION INDICATOR IN SUBROUTINE TARGET =1 STOP ON TIME =2 STOP AT SPHERE OF INFLUENCE =3 STOP AT CLOSEST APPROACH
NOPAR	NUMBER OF TARGET PARAMETERS FOR CURRENT EVENT
KWIT	TERMINATION FLAG =0 CONTINUE RUN =1 TERMINATE RUN
IPRE	CASE FLAG =0 FIRST CASE =1 STACKED CASE
NCPR	NUMBER OF INTEGRATION INCREMENTS BETWEEN PRINT-OUTS OF NOMINAL TRAJECTORY
IFINT(10)	NOT USED IN THIS TRAJECTORY VERSION
KGYD(10)	INDICES OF EVENTS TO BE PROCESSED
KSICA	FLAG INDICATING STAGE OF NOMINAL TRAJECTORY =1 SOI NOT YET INTERSECTED =2 SOI INTERSECTED BUT NO CLOSEST APPROACH =3 CLOSEST APPROACH ALREADY ENCOUNTERED
KUR	INDEX OF CURRENT EVENT
KAXTAR(3)	KEY DEFINING AUXILIARY PARAMETERS FOR CURRENT EVENT
LVLS(10)	NUMBER OF ACCURACY LEVELS TO BE USED ON EACH TARGETING EVENT
NOSOI	OUTER TARGETING FLAG =0 NORMAL TARGETING =1 OUTER TARGETING

/TARVAR/ MODE ERRAN, SIMUL

XTAR(6,10)	DESIRED TARGET VALUES
XTOL(6,10)	TOLERANCES ON TARGET PARAMETERS
XAC(5,10)	ACCURACY LEVELS EMPLOYED IN TARGETING
XPERV(10)	VELOCITY PERTURBATION USED TO COMPUTE TARGETING MATRIX
XDVMAX(10)	MAXIMUM ALLOWABLE VELOCITY CORRECTION
XFAC(10)	SPHERE OF INFLUENCE FACTORS
XDELV(3,10)	NONLINEAR VELOCITY CORRECTION
TGT3(10)	DESIRED TARGET TIMES REFERENCED TO INITIAL TRAJECTORY TIME
LKTAR(6,10)	ARRAY DEFINING TARGET PARAMETERS
LKTP(10)	ARRAY OF TARGET PLANETS
LKLP(10)	ARRAY OF LAUNCH PLANETS
LNPAR(10)	NUMBER OF TARGET PARAMETERS DESIRED
LLVLS(10)	NUMBER OF INTEGRATION ACCURACY LEVLES USED

/TIM / MODE: ERRAN, SIMUL

DATEJ	JULIAN DATE OF INITIAL TRAJECTORY TIME (REFERENCED TO 1950)
TRTM1	INITIAL TRAJECTORY TIME
DELTM	TIME INCREMENT
FNTM	FINAL TRAJECTORY TIME
UNIVT	UNIVERSAL TIME
TRTMB	TRAJECTORY TIME AT BEGINNING OF TRAJECTORY

/TMW2 / MODE: SIMUL

T1	EIGENVECTOR EVENT TIMES
T2	PREDICTION EVENT STARTING TIMES
T4	CONIC COMPUTATION EVENT TIMES
T5	QUASI-LINEAR EVENT TIMES
T6	NOT USED
T7	NOT USED

/TRAJCO/ MODE: ERRAN, SIMUL

DTMAX	MAXIMUM TIME INCREMENT FOR WHICH ISTMC IS VALID
ACCND	ACCURACY USED IN NUMERICAL DIFFERENCING IF NDACC INDICATES
DTSUN	STATE TRANSITION INTEGRATION INTERVAL WHEN THE SUN IS CENTRAL BODY AND ISTM1=1
DTPLAN	STATE TRANSITION INTEGRATION INTERVAL WHEN TARGET PLANET IS CENTRAL BODY AND ISTM1=1
NTMC	NOMINAL TRAJECTORY CODE
ISTMC	STATE TRANSITION MATRIX CODE
ISTM1	ALTERNATE STATE TRANSITION MATRIX CODE
NDACC	NUMERICAL DIFFERENCING ACCURACY CODE

/TRJ / MODE: ERRAN, SIMUL

RCA1(6)	STATE AT CLOSEST APPROACH ON ORIGINAL NOMINAL
RCA2(6)	STATE AT CLOSEST APPROACH ON MOST RECENT NOMINAL
RCA3(6)	STATE AT CLOSEST APPROACH ON ACTUAL TRAJECTORY
RSOI1(3)	POSITION AT SPHERE OF INFLUENCE ON ORIGINAL NOMINAL
RSOI2(3)	POSITION AT SPHERE OF INFLUENCE ON MOST RECENT NOMINAL
RSOI3(3)	POSITION AT SPHERE OF INFLUENCE ON ACTUAL TRAJECTORY

VSOI1(3)	VELOCITY AT SPHERE OF INFLUENCE ON ORIGINAL NOMINAL
VSOI2(3)	VELOCITY AT SPHERE OF INFLUENCE ON MOST RECENT NOMINAL
VSOI3(3)	VELOCITY AT SPHERE OF INFLUENCE ON ACTUAL TRAJECTORY
TCA1	TIME AT CLOSEST APPROACH OF ORIGINAL NOMINAL
TCA2	TIME AT CLOSEST APPROACH OF MOST RECENT NOMINAL
TCA3	TIME AT CLOSEST APPROACH OF ACTUAL TRAJECTORY
TSOI1	TIME AT SPHERE OF INFLUENCE OF ORIGINAL NOMINAL
TSOI2	TIME AT SPHERE OF INFLUENCE OF MOST RECENT NOMINAL
TSOI3	TIME AT SPHERE OF INFLUENCE OF ACTUAL TRAJECTORY
BSI1	B ON ORIGINAL NOMINAL
BSI2	B ON MOST RECENT NOMINAL
BSI3	B ON ACTUAL TRAJECTORY
BDTSI1	B DOT T ON ORIGINAL NOMINAL
BDTSI2	B DOT T ON MOST RECENT NOMINAL
BDTSI3	B DOT T ON ACTUAL TRAJECTORY
BDRSI1	B DOT R ON ORIGINAL NOMINAL
BDRSI2	B DOT R ON MOST RECENT NOMINAL
BDRSI3	B DOT R ON ACTUAL TRAJECTORY
ISOI1	SPHERE OF INFLUENCE CODE FOR ORIGINAL NOMINAL
ISOI2	SPHERE OF INFLUENCE CODE FOR MOST RECENT NOMINAL

ISOI3	SPHERE OF INFLUENCE CODE FOR ACTUAL TRAJECTORY
ICA1	CLOSEST APPROACH CODE FOR ORIGINAL NOMINAL
ICA2	CLOSEST APPROACH CODE FOR MOST RECENT NOMINAL
ICA3	CLOSEST APPROACH CODE FOR ACTUAL TRAJECTORY

/TMTRIX/ MODE: NOMNAL, ERRAN, SIMUL

CHI(3,3)	SENSITIVITY MATRIX (TRANSFERRED FOR OUTPUT)

/VM / MODE: NOHNAL, ERRAN, SIMUL

ALNGTH	LENGTH UNITS PER A.U.
TM	TIME UNITS PER DAY
DELTP	PRINT INCREMENTS (IN DAYS)
RC(6)	STATE AT CLOSEST APPROACH
DC	JULIAN DATE,EPOCH 1900,AT CLOSEST APPROACH
RSI(3)	POSITION AT SPHERE OF INFLUENCE
VSI(3)	VELOCITY AT SPHERE OF INFLUENCE
DSI	JULIAN DATE, EPOCH 1900, AT SPHERE OF INFLUENCE
RVS(6)	POSITION OF VEHICLE RELATIVE TO VIRTUAL MASS
VMU	GRAVITATIONAL CONSTANT OF VIRTUAL MASS
B	B AT SPHERE OF INFLUENCE
BDT	B DOT T
BDR	B DOT R
DELTH	INCREMENT IN TRUE ANOMALY USED
TIMINT	TOTAL TIME USED
RE(6)	POSITION AND VELOCITY OF EARTH
RTP(6)	POSITION AND VELOCITY OF TARGET PLANET
CAINC	INCLINATION AT CLOSEST APPROACH
RCA	MAGNITUDE OF CLOSEST APPROACH POSITION VECTOR
TACA	TRAJECTORY SEMIMAJOR AXIS WITH RESPECT TO TARGET BODY AT CLOSEST APPROACH TO TARGET BODY

SSS(3)	DIRECTION COSINE VECTOR OF SPACECRAFT SPIN AXIS
NLP	CODE OF LAUNCH PLANET
NBOD	NUMBER OF BODIES USED IN VIRTUAL MASS PROGRAM
NB(11)	CODES OF PLANETS
NTP	CODE OF TARGET PLANET
INPR	PRINT INCREMENTS (IN INCREMENTS)
IPROB	PROBLEM NUMBER
ISPH	SPHERE OF INFLUENCE CODE =0 SPHERE OF INFLUENCE NOT INTERSECTED =1 SPHERE OF INFLUENCE ALREADY ENCOUNTERED
INCMT	TOTAL INCREMENTS USED
IEPHEM	EPHEMERIS CODE
ICL	CLOSEST APPROACH CODE =0 CLOSEST APPROACH NOT ENCOUNTERED =1 CLOSEST APPROACH ALREADY ENCOUNTERED
IPRINT	PRINT CODE =0 OUTPUT INITIAL AND FINAL DATA =1 DO NOT OUTPUT INITIAL AND FINAL DATA
ICL2	CLOSEST APPROACH TERMINATION CODE =0 DO NOT STOP AT CLOSEST APPROACH =1 STOP AT CLOSEST APPROACH

/XXXL / MODE: ERRAN, SIMUL

XSL(24)	SOLVE-FOR PARAMETER LABELS
XU(8)	DYNAMIC CONSIDER PARAMETER LABELS
XV(15)	MEASUREMENT CONSIDER PARAMETER LABELS
X'AB(6)	VEHICLE POSITION/ VELOCITY VECTOR COMPONENT NAMES
XNM(24)	AUGUMENTATION PARAMETER LABELS
KPRINT	CORRELATION MATRIX PRINT CODE

/ZERDAT/ MODE: NOMNAL

ZDAT(6)	ZERO ITERATE VECTOR
RP	PARKING ORBIT RADIUS
FI	INJECTION TRUE ANOMALY
PSI1	ANGLE OF FIRST BURN
PSI2	ANGLE OF SECOND BURN
TIM1	TIME INTERVAL OF FIRST BURN
TIM2	TIME INTERVAL OF SECOND BURN
THELS	LONGITUDE OF LAUNCH SITE
PHILS	LATITUDE OF LAUNCH SITE
TI	NOT USED
TF	NOT USED
THEDOT	ROTATION RATE OF LAUNCH PLANET
RPRAT	PARKING ORBIT INVERSE RATE

SIGMAL	NOMINAL LAUNCH AZIMUTH
IZERO	ZERO ITERATION FLAG =0 INITIAL STATE READ IN =1 PLANET-TO-PLANET =2 PLANET-TO-POINT =3 POINT-TO-PLANET =4 POINT-TO-POINT =10 LUNAR TARGETING
KOAST	PARKING ORBIT INDICATOR =-1 SHORT COAST =1 LONG COAST
LTARG	TYPE OF MISSION FOR TARGETING =0 INTERPLANETARY MISSION =1 LUNAR MISSION

/ZOUT / MODE: NOMNAL

VHPM	MAGNITUDE OF HYPERBOLIC EXCESS VELOCITY AT TARGET BODY (VHP VECTOR)
DPA	DECLINATION OF VHP
RAP	RIGHT ASCENSION OF VHP

5.2 COMMON VARIABLES IN ALPHABETICAL ORDER

IN THIS SECTION ALL VARIABLES APPEARING IN COMMON ARE LISTED AND DEFINED IN ALPHABETICAL ORDER. THE SECOND FIELD SERVES TO IDENTIFY THE BLOCK IN WHICH THE VARIABLE APPEARS.

A(2,3)	PBLK	FTA IMPACT PLANE TRANSFORMATION MATRIX
AALP(20)	SIMCNT	ACTUAL ERROR IN POINTING ANGLE 1
AAUX(3)	TAREAL	ACTUAL AUXILIARY VALUES OF ITERATE
ABET(20)	SIMCNT	ACTUAL ERROR IN POINTING ANGLE 2
AC(5,10)	TAREAL	ACCURACY LEVELS(UP TO 5) USED IN EACH GUIDANCE EVENT
ACC	MISC	ACCURACY FIGURE USED IN VIRTUAL MASS PROGRAM
ACC1	SIM2	ACCURACY USED IN ACTUAL TRAJECTORY
ACCND	TRAJCD	ACCURACY USED IN NUMERICAL DIFFERENCING IF NDACC INDICATES
ACKT	TAREAL	TRAJECTORY INTEGRATION ACCURACY
ADA(3,6)	BAIM	VARIATION MATRIX
ADEVSB(24)	SIM1	ACTUAL DEVIATION IN SOLVE-FOR PARAMETERS AT TRAJECTORY BEGINNING
ADEVX(6)	SIM1	ACTUAL DEVIATION IN THE STATE VECTOR
ADEVXB(6)	SIM1	ACTUAL DEVIATION IN STATE VECTOR AT BEGINNING OF TRAJECTORY
ADEVXS(24)	SIM1	ACTUAL DEVIATION IN SOLVE-FOR PARAMETERS
AINC7	EVENT	ORBIT INSERTION VARIABLES. NON-FUNCTIONAL IN IN EXISTING PROGRAM
AK(6,4)	STM	KALMAN GAIN CONSTANT FOR POSITION/VELOCITY STATE

AL(4,15)	STM	OBSERVATION MATRIX RELATING OBSERVABLES TO MEASUREMENT CONSIDER PARAMETER STATE
ALNGTH	VM	LENGTH UNITS PER A.U.
AM(4,24)	STM	OBSERVATION MATRIX RELATING OBSERVABLES TO SOLVE-FOR PARAMETER STATE
ANODE7	EVENT	ORBIT INSERTION VARIABLES. NON-FUNCTIONAL IN IN EXISTING PROGRAM
ANG	IMPTAR	TARGET INCLINATION CONVERTED FROM INPUT FORMAT TO VALUE BETWEEN 0 AND 180 DEGREES AND SATISFYING APPROACH ASYMPTOTE CONSTRAINT
ANOIS(4)	SIM1	ACTUAL WHITE NOISE
APRO(20)	SIMCNT	ACTUAL PROPORTIONALITY ERROR
AR(4,4)	SIM1	ACTUAL MEASUREMENT NOISE
ARES(20)	SIMCNT	ACTUAL RESOLUTION ERROR
ATAR(3)	TAREAL	ACTUAL TARGET VALUES OF ITERATE
ATRANS(6)	BAIM	CLOSEST APPROACH STATE
AVARM(12)	SIMCNT	VARIANCE OF ACTUAL MEASUREMENT NOISE
AY(4)	SIM1	ACTUAL MEASUR[illegible]NT
AYMEY(4)	SIM1	ACTUAL MEASUREMENT MINUS ESTIMATED MEASUREMENT
B	VM	B AT SPHERE OF INFLUENCE
BCON(3)	LUNART	MAXIMUM STEP SIZES OF CONTROLS
BDR	VM	B DOT R
BDT	VM	B DOT T
BIA(12)	MISC	MEASUREMENT BIASES

BSI1	TRJ	B ON ORIGINAL NOMINAL
BSI2	TRJ	B ON MOST RECENT NOMINAL
BSI3	TRJ	B ON ACTUAL TRAJECTORY
BDTSI1	TRJ	B DOT T ON ORIGINAL NOMINAL
BDTSI2	TRJ	B DOT T ON MOST RECENT NOMINAL
BDTSI3	TRJ	B DOT T ON ACTUAL TRAJECTORY
BDRSI1	TRJ	B DOT R ON ORIGINAL NOMINAL
BDRSI2	TRJ	B DOT R ON MOST RECENT NOMINAL
BDRSI3	TRJ	B DOT R ON ACTUAL TRAJECTORY
CAI	LUNART	DESIRED CLOSEST APPROACH EQUATORIAL INCLINATION
CAINC	VH	INCLINATION AT CLOSEST APPROACH
CHI(3,3)	TMTRX	SENSITIVITY MATRIX(TRANSFERRED FOR OUTPUT)
CMPNM(30)	NAME	COMPONENT NAME
CN(80)	BLK	CONSTANTS USED TO CALCULATE THE ORBITAL ELEMENTS OF THE FIRST FIVE PLANETS (SEE LARGE ARRAY DEFNS IN SECTION 5.3)
CR	PBLK	CAPTURE RADIUS OF TARGET PLANET
CTOL(6)	TAREAL	TOLERANCES FOR CURRENT EVENT
CXSU(24,8)	STM	CORRELATION BETWEEN SOLVE-FOR PARAMETERS AND DYNAMIC CONSIDER PARAMETERS
CXSUB(24,8)	STM	CORRELATION BETWEEN SOLVE-FOR PARAMETERS AND DYNAMIC CONSIDER PARAMETERS AT INITIAL TIME
CXSUG(24,8)	GUI	CONTROL CORRELATION BETWEEN SOLVE-FOR PARAMETERS AND DYNAMIC CONSIDER PARAMETERS
CXSUP(24,8)	GUI	CORRELATION BETWEEN SOLVE-FOR PARAMETERS AND DYNAMIC CONSIDER PARAMETERS PRIOR TO PROCESSING A MEASUREMENT

CXSV(24,15)	STM	CORRELATION BETWEEN SOLVE-FOR PARAMETERS AND MEASUREMENT CONSIDER PARAMETERS
CXSVB(24,15)	STM	CORRELATION BETWEEN SOLVE-FOR PARAMETERS AND MEASUREMENT CONSIDER PARAMETERS AT INITIAL TIME
CXSVG(24,15)	GUI	CONTROL CORRELATION BETWEEN SOLVE-FOR PARAMETERS AND MEASUREMENT CONSIDER PARAMETERS
CXSVP(24,15)	STM	CORRELATION BETWEEN SOLVE-FOR PARAMETERS AND MEASUREMENT CONSIDER PARAMETERS PRIOR TO PROCESSING A MEASUREMENT
CXU(6,8)	STM	CORRELATION BETWEEN POSITION/VELOCITY STATE AND DYNAMIC CONSIDER PARAMETERS
CXUB(6,8)	STM	CORRELATION BETWEEN POSITION/VELOCITY STATE AND DYNAMIC CONSIDER PARAMETERS AT INITIAL TIME
CXUG(6,8)	GUI	CONTROL CORRELATION BETWEEN POSITION/VELOCITY STATE AND DYNAMIC CONSIDER PARAMETERS
CXUP(6,8)	STM	CORRELATION BETWEEN POSITION/VELOCITY STATE AND DYNAMIC CONSIDER PARAMETERS PRIOR TO PROCESSING A MEASUREMENT
CXV(6,15)	STM	CORRELATION BETWEEN POSITION/VELOCITY STATE AND MEASUREMENT CONSIDER PARAMETERS
CXVB(6,15)	STM	CORRELATION BETWEEN POSITION/VELOCITY STATE AND MEASUREMENT CONSIDER PARMETERS AT INITIAL TIME
CXVG(6,15)	GUI	CONTROL CORRELATION BETWEEN POSITION/VELOCITY STATE AND MEASUREMENT CONSIDER PARAMETERS
CXVP(6,15)	STM	CORRELATION BETWEEN POSITION/VELOCITY STATE AND MEASUREMENT CONSIDER PARAMETERS PRIOR TO PROCESSING A MEASUREMENT

CXXS(6,24)	STM	CORRELATION BETWEEN POSITION/VELOCITY STATE AND SOLVE-FOR PARAMETERS
CXXSB(6,24)	STM	CORRELATION BETWEEN POSITION/VELOCITY STATE AND SOLVE-FOR PARAMETERS AT INITIAL TIME
CXXSG(6,24)	GUI	CONTROL CORRELATION BETWEEN POSITION/ VELOCITY STATE AND SOLVE-FOR PARAMETERS
CXXSP(6,24)	STM	CORRELATION BETWEEN POSITION/VELOCITY STATE AND SOLVE-FOR PARAMETERS PRIOR TO PROCESSING A MEASUREMENT
D1	TAREAL	JULIAN DATE ASSOCIATED WITH RIN ARRAY
DAB	SIMCNT	BIAS IN SEMI-MAJOR AXIS OF TARGET PLANET
DATEJ	TIME	JULIAN DATE OF INITIAL TRAJECTORY TIME (REFERENCED TO 1950)
DAUX(3)	TAREAL	DESIRED AUXILIARY PARAMETER VALUES OF ITERATE
DC	VM	JULIAN DATE,EPOCH 1900,AT CLOSEST APPROACH
DEB	SIMCNT	BIAS IN ECCENTRICITY OF TARGET PLANET
DECLIN	LUNART	DECLINATION OF APPROACH ASYMPTOTE WITH RESPECT TO LUNAR EQUATOR
DELAXS	CONST3	TARGET PLANET SEMI-MAJOR AXIS FACTOR USED IN NUMERICAL DIFFERENCING
DELECC	CONST3	TARGET PLANET ECCENTRICITY FACTOR USED IN NUMERICAL DIFFERENCING
DELICL	CONST3	TARGET PLANET INCLINATION FACTOR USED IN NUMERICAL DIFFERENCING
DELMA	CONST3	TARGET PLANET MEAN ANOMALY FACTOR USED IN NUMERICAL DIFFERENCING
DELMUP	CONST3	TARGET PLANET GRAVITATIONAL CONSTANT FACTOR USED IN NUMERICAL DIFFERENCING
DELMUS	CONST3	SUN GRAVITATIONAL CONSTANT FACTOR USED IN NUMERICAL DIFFERENCING

DELNOD	CONST3	TARGET PLANET LONGITUDE OF THE ASCENDING NODE FACTOR USED IN NUMERICAL DIFFERENCING .
DELPX(3)	EXE	VELOCITY CORRECTION TO BE MODELED AS AN IMPULSE SERIES
DELTAT	TAREAL	NUMBER OF DAYS INTEGRATION IS TO CONTINUE IF NO OTHER STOPPING CONDITION OCCURS
DELTAV(3)	TAREAL	CORRECTIONS TO BE ADDED TO VELOCITY COMPONENTS FOR NEXT ITERATION
DELTH	VM	INCREMENT IN TRUE ANOMALY USED
DELTM	TIME	TIME INCREMENT
DELTP	VM	PRINT INCREMENTS (IN DAYS)
DELV(3,10)	TAREAL	EXTERNALLY SUPPLIED VELOCITY CORRECTION OR VELOCITY INCREMENT COMPUTED BY INSERTION
DELV(3,10)	PBLK	ARRAY OF EXTERNALLY-SUPPLIED VELOCITY CHANGES
DELW	CONST3	TARGET PLANET ARGUMENT OF PERIAPSIS FACTOR USED IN NUMERICAL DIFFERENCING
DG(10)	TAREAL	JULIAN DATES OF EVENT TIMES
DIB	SIMCNT	BIAS IN INCLINATION OF TARGET PLANET
DINTG(10)	TAREAL	NOT USED IN CURRENT TARGET VERSION
DIPX	EXE	JULIAN DATE TRANSFERRED TO EXCUT OR EXCUTS
DMAB	SIMCNT	BIAS IN MEAN ANOMALY
DMUPB	SIMCNT	BIAS IN GRAVITATIONAL CONSTANT OF TARGET PLANET
DMUSB	SIMCNT	BIAS IN GRAVITATIONAL CONSTANT OF SUN
DNCN(3)	CONST	CONSTANTS FROM WHICH DYNAMIC NOISE IS COMPUTED
DNOB	SIMCNT	BIAS IN LONGITUDE OF ASCENDING NODE
DPA	ZOUT	DECLINATION OF VHP

DSI	VM	JULIAN DATE, EPOCH 1900, AT SPHERE OF INFLUENCE
DT(10)	TAREAL	JULIAN DATES OF TARGET TIMES
DTAR(3)	TAREAL	DESIRED TARGET VALUES OF ITERATE
DTAR(3)	LUNART	TARGET VALUES OF SMA,B.T, AND B.R IN LUNAR TARGETING
DTI	PULS	TIME INTERVAL (DAYS) BETWEEN SUCCESSIVE PULSES
DTIME	OVERL	TIME INTERVAL BETWEEN ORBITAL INSERTION DECISION AND EXECUTION
DTMAX	TRAJCD	MAXIMUM TIME INCREMENT FOR WHICH ISTMC IS VALID
DTPLAN	TRAJCD	STATE TRANSITION INTEGRATION INTERVAL WHEN TARGET PLANET IS CENTRAL BODY AND ISTM1=1
DTSUN	TRAJCD	STATE TRANSITION INTEGRATION INTERVAL WHEN THE SUN IS CENTRAL BODY AND ISTM1=1
DUMMYQ(4)	EXE	ARRAY OF EXECUTION ERROR VARIANCES
DUR	PULS	DURATION OF SINGLE PULSE
DV8(3)	EVENT	ORBIT INSERTION VARIABLE. NON-FUNCTIONAL IN EXISTING PROGRAM
DVF(3)	PULS	VELOCITY INCREMENT ADDED ON FINAL PULSE
DVI(3)	PULS	VELOCITY INCREMENT ADDED ON TYPICAL PULSE
DVMAX(10)	TAREAL	MAXIMUM ALLOWABLE CHANGE IN ANY VELOCITY COMPONENT FOR EACH EVENT
DVN(3)	PBLK	COMMANDED VELOCITY CORRECTION TRANSFERRED TO BIAIM
DVRB(3)	PBLK	VELOCITY CORRECTION REQUIRED TO REMOVE AIMPOINT BIAS

DVUP(3)	PBLK	UPDATE VELOCITY CORRECTION
DWB	SIMCNT	BIAS IN ARGUMENT OF PERIAPSIS
ECC7	EVENT	ORBIT INSERTION VARIABLES. NON-FUNCTIONAL IN IN EXISTING PROGRAM
EDEVSM(24)	SIM1	ESTIMATED DEVIATION IN SOLVE-FOR PARAMETERS (FOR ADAPTIVE FILTERING)
EDEVX(6)	SIM1	ESTIMATED DEVIATION IN THE STATE VECTOR
EDEVXM(6)	SIM1	ESTIMATED DEVIATION IN THE STATE VECTOR (FOR ADAPTIVE FILTERING)
EDEVXS(24)	SIM1	ESTIMATED DEVIATION IN SOLVE-FOR PARAMETERS
EIGHT	DPNUM	THE NUMBER EIGHT (8) TO NINE SIGNIFICENT FIGURES. TARGETING MODE ONLY
ELMNT(80)	BLK	CONTAINS THE ORBITAL ELEMENTS OF THE PLANETS (SEE LARGE ARRAY DEFNS IN SECTION 5.3)
EM(2,6)	GUI	VARIATION MATRIX RELATING POSITION /VELOCITY DEVIATIONS TO B.T AND B.R DEVIATIONS
EM1	DPNUM	THE NUMBER 1.E-1 TO NINE SIGNIFICANT FIGURES
EM2	DPNUM	THE NUMBER 1.E-2 TO NINE SIGNIFICANT FIGURES
EM3	DPNUM	THE NUMBER 1.E-3 TO NINE SIGNIFICANT FIGURES
EM4	DPNUM	THE NUMBER 1.E-4 TO NINE SIGNIFICANT FIGURES
EM5	DPNUM	THE NUMBER 1.E-5 TO NINE SIGNIFICANT FIGURES
EM6	DPNUM	THE NUMBER 1.E-6 TO NINE SIGNIFICANT FIGURES

EM7	DPNUM	THE NUMBER 1.E-7 TO NINE SIGNIFICANT FIGURES
EM8	DPNUM	THE NUMBER 1.E-8 TO NINE SIGNIFICANT FIGURES
EM9	DPNUM	THE NUMBER 1.E-9 TO NINE SIGNIFICANT FIGURES
EM13	DPNUM	THE NUMBER 1.E-13 TO NINE SIGNIFICANT FIGURES
EM50	DPNUM	THE NUMBER 1.E-50 TO NINE SIGNIFICANT FIGURES
EMN(15)	BLK	THE CONSTANTS USED TO CALCULATE THE ORBITAL ELEMENTS OF THE MOON
EMU	LUNART	GRAVITATIONAL CONSTANT OF EARTH(KM3/SEC2)
EPS	CONST	OBLIQUITY OF EARTH
EQECP(3,3)	TAREAL	TRANSFORMATION FROM ECLIPTIC TO EQUATORIAL SYSTEM FOR TARGET PLANE
EQLQ(3,3)	LUNART	TRANSFORMATION MATRIX FROM EARTH-EQUATORIAL TO LUNAR EQUATORIAL COORDINTES
EVNM(11)	NAME	EVENT NAME
EXEC(3,3)	PBLK	EXECUTION ERROR COVARIANCE MATRIX
EY(4)	SIM1	ESTIMATED MEASUREMENT
F(44,4)	COM	CONTAINS THE POSITIONS AND VELOCITIES OF THE PLANETS AT A SPECIFIED TIME PLUS THE POSITIONS AND VELOCITIES OF THE SPACECRAFT RELATIVE TO THE PLANETS
FAC(3)	TAREAL	SCALING FACTORS USED IN BAD STEP CHECK
FACP	MISC	POSITION FACTOR USED IN NUMERICAL DIFFERENCING
FACV	MISC	VELOCITY FACTOR USED IN NUMERICAL DIFFERENCING
FI	ZERDAT	INJECTION TRUE ANOMALY

FIVE	DPNUM	THE NUMBER FIVE (5) TO NINE SIGNIFICANT FIGURES. TARGETING MODE ONLY
FNTM	TIME	FINAL TRAJECTORY TIME
FOP	CONST2	OFF-DIAGONAL ANNIHILATION VALUE FOR POSITION EIGENVALUES
FOUR	DPNUM	THE NUMBER FOUR (4) TO NINE SIGNIFICANT FIGURES. TARGETING MODE ONLY
FOV	CONST2	OFF-DIAGONAL ANNIHILATION VALUE FOR VELOCITY EIGENVALUES
FS(2,5)	PULS	F-SERIES COEFFICIENTS OF LAUNCH AND TARGET BODIES
G(4,8)	STM	OBSERVATION MATRIX RELATING OBSERVABLES TO DYNAMIC CONSIDER PARAMETER STATE
GA(3,6)	OVERZ	GUIDANCE MATRIX
GG(3)	PULS	GRAVITIONAL CONSTANTS OF SUN, LAUNCH, AND TARGET BODIES
GS(2,4)	PULS	G-SERIES COEFFIEIENTS OF LAUNCH AND TARGET BODIES
H(4,6)	STM	OBSERVATION MATRIX RELATING OBSERVABLES TO POSITION/VELOCITY STATE
HALF	DPNUM	THE NUMBER ONE-HALF (1/2) TO NINE SIGNIFICANT FIGURES
HP7	EVENT	ORBIT INSERTION VARIABLES. NON-FUNCTIONAL IN EXISTING PROGRAM
HPHR(4,4)	STM	NON-FUNCTIONAL IN PRESENT ERROR ANALYSIS PROGRAM
IAMNF	SIMCNT	ACTUAL MEASUREMENT NOISE FLAG
IAUGIN(24)	STVEC	INPUT AUGMENTATION VECTOR OF ONE*S AND ZERO-S
IAUG(24)	STVEC	AUGMENTATION VECTOR
IAUGDC(8)	STVEC	DYNAMIC CONSIDER AUGMENTATION VECTOR

IAUGMC(15)	STVEC	MEASUREMENT CONSIDER AUGMENTATION VECTOR
IBAD	TARINT	BAD STEP FLAG FOR CURRENT ACCURACY LEVEL =1, DO NOT CHECK FOR BAD STEP =2, CHECK FOR BAD STEP
IBADS(10)	TARINT	BAD STEP FLAGS FOR EACH TARGETING EVENT =1 NEVER USE BAD STEP CHEQUE =2 USE BAD STEP CHECK AT FINAL LEVEL ONLY =3 USE BAD STEP CHECK AT ALL LEVELS
IBAG	BAIM	NOT USED
IBARY	CNTRIC	REFERENCE COORDINATE SYSTEM CODE
IBAST	TARINT	BAD STEP CHECK INDICATOR FOR CURRENT EVENT
IBIAS	PBLK	BIASED AIMPOINT GUIDANCE EVENT FLAG = 0 AIMPOINT NOT BIASED =1 AIMPOINT BIASED
ICA1	TRJ	CLOSEST APPROACH CODE FOR ORIGINAL NOMINAL
ICA2	TRJ	CLOSEST APPROACH CODE FOR MOST RECENT NOMINAL
ICA3	TRJ	CLOSEST APPROACH CODE FOR ACTUAL TRAJECTORY
ICDQ3(20)	EVENT	ARRAY OF CODES WHICH DETERMINE WHICH EXECUTION POLICIES ARE TO BE USED IN GUIDANCE EVENTS
ICDT3(20)	EVENT	CODES WHICH DETERMINE WHICH GUIDANCE POLICIES ARE BEING USED
ICL	VM	CLOSEST APPROACH CODE =0 CLOSEST APPROACH NOT ENCOUNTERED =1 CLOSEST APPROACH ENCOUNTERED
ICL2	VM	CLOSEST APPROACH TERMINATION CODE =0 DO NOT STOP AT CLOSEST APPROACH =1 STOP AT CLOSEST APPROACH
ICODE	OVER1	MEASUREMENT CODE
ICOOR	MISC	CODE TO DETERMINE WHICH COORDINATE SYSTEM THE INITIAL STATE VECTOR IS INPUT

ICOORD	CNTRIC	NON-FUNCTIONAL IN ERROR ANALYSIS MODE
IDENS	PBLK	PROBABILITY DENSITY FUNCTION CODE. NON-FUNCTIONAL
IDNF	MISC	DYNAMIC NOISE FLAG
IEIG	EVENT	CODE USED TO DECIDE IF BOTH POSITION AND VELOCITY EIGENVECTORS ARE REQUESTED
IEPHEM	VM	EPHEMERIS CODE
IEVNT(50)	EVENT	CODES OF EVENTS
IFINT(10)	TARINT	NOT USED IN THIS TRAJECTORY VERSION
IGP	OVERZ	MIDCOURSE GIUDANCE POLICY CODE
IGUID(5,10)	BAIM	ARRAY OF GUIDANCE EVENT CODES
IHYP1	EVENT	HYPERELLIPSOID CODE USED TO DETERMINE IF K=1, K=3, OR BOTH
II	BAIM	GUIDANCE EVENT COUNTER
IIGP	PBLK	MIDCOURSE GUIDANCE POLICY CODE
IIPOL	EVENT	CODE WHICH DETERMINES IF EITHER TWO-VARIBLE OR THREE-VARIABLE B-PLANE GUIDANCE POLICY HAS OCCURRED
IMNF	MISC	MEASUREMENT NOISE FLAG
INC	COM	DETERMINE WHETHER THE ABOVE OPTION IS TO BE USED
INCMNT	COM	NUMBER OF INCREMENTS USED
INCMT	VM	TOTAL INCREMENTS USED
INCPR	COM	SPECIFIES AFTER HOW MANY TIME INCREMENTS PRINT-OUT IS TO OCCUR
INITIAL	CNTRIC	NON-FUNCTIONAL IN ERROR ANALYSIS MODE
INPR	VM	PRINT INCREMENTS (IN INCREMENTS)
INPX	EXE	IMP[illegible] SERIES CODE

IOPT7	EVENT	ORBIT INSERTION VARIABLE. NON-FUNCTIONAL IN EXISTING PROGRAMS
IPG	COM	PAGE NUMBER
IPHASE	TARINT	PHASE COUNTER FOR CURRENT TARGETING EVENT
IPOL	EVENT	CODE WHICH DETERMINES IF FIXED-TIME-OF-ARRIVAL GUIDANCE EVENT HAS OCCURED
IPQ	BAIM	NOT USED
IPR	COM	A CODE WHICH DETERMINES IF PRINT-OUT IS TO OCCUR AFTER A SPECIFIED NUMBER OF DAYS
IPRE	PRINT	CONTROLS INITIALIZATION OF PROGRAM CONSTANTS IN SUBROUTINE -PRELIM-.
IPRINT	VM	PRINT CODE =0 OUTPUT INITIAL AND FINAL DATA =1 DO NOT OUTPUT DATA
IPROB	VM	PROBLEM NUMBER
IPRT(4)	COM	SPECIFIES PRINT OPTIONS (NOT APPLICABLE TO STEAP TRAJECTORY) .
ISOI1	TRJ	SPHERE OF INFLUENCE CODE FOR ORIGINAL NOMINAL
ISOI2	TRJ	SPHERE OF INFLUENCE CODE FOR MOST RECENT NOMINAL
ISOI3	TRJ	SPHERE OF INFLUENCE CODE FOR ACTUAL TRAJECTORY
ISP2	MISC	SPHERE OF INFLUENCE FLAG
ISPH	VM	SPHERE OF INFLUENCE CODE =0 SPHERE OF INFLUENCE NOT INTERSECTED =1 SPHERE OF INFLUENCE INTERSECTED
ISTART	TARINT	STAGE OF INITIAL TARGETING =0 NO TARGETING STARTED =1 FIRST PHASE STARTED -HAVE TARG MATRIX =2 SECOND PHASE STARTED - HAVE TARG MATRIX

ISTM1	TRAJCD	ALTERNATE STATE TRANSITION MATRIX CODE
ISTMC	TRAJCD	STATE TRANSITION MATRIX CODE
ISTOP	TARINT	STOPPING CONDITION INDICATOR IN SUBROUTINE TARGET =1, STOP ON TIME =2, STOP AT SPHERE-OF-INFLUENCE =3, STOP AT CLOSEST APPROACH
ITAG	LUNART	FLAG SPECIFYING STAGE OF TARGETING =1 IN SMA TARGETING =2 IN SMA, INC, RCA TARGETING
ITARM	TARINT	FLAG TO CONTROL CONSTRUCTION OF TARGETING MATRICES =0, DO NOT CALCULATE STATE TRANSITION =1, CALCULATE STATE TRANSITION MATRIX AFTER EACH ITERATION
ITR	MISC	MODE FLAG
ITRAT	COM	IN INTERNAL CODE USED TO DETERMINE HOW MANY ITERATIONS HAVE BEEN ACCOMPLISHED IN THE VIRTUAL MASS PROCEDURE
IX	OVERX	NONLINEAR GUIDANCE CODE
IZERO	ZERDAT	ZERO ITERATION FLAG =0 INITIAL STATE READ IN =1 PLANET TO PLANET =2 PLANET TO POINT =3 POINT TO PLANET =4 POINT TO POINT =10 LUNAR TARGETING
JX	OVERX	GUIDANCE EVENT COUNTER
KAXTAR(3)	TARINT	KEY DEFINING AUXILIARY PARAMETERS FOR CURRENT EVENT
KEYTAR(3)	TARINT	KEY DEFINING DESIRED TARGET PARAMETERS FOR CURRENT EVENT
KGYD(10)	TARINT	FLAG INDICATING GUIDANCE INFORMATION IN CORRESPONDING COLUMNS OF INPUT ARRAYS =1, INFORMATION =0, NO INFORMATION

KL	COM	PROBLEM NUMBER (NOMNAL ONLY)
KMXQ(10)	TARINT	COMPUTE/EXECUTE MODES FOR EACH EVENT =1 COMPUTE VELOCITY CORRECTION ONLY =2 EXECUTE VELOCITY CORRECTION ONLY =3 COMPUTE AND IMMEDIATELY EXECUTE CORRECTIONS =4 COMPUTE BUT EXECUTE CORRECTION LATER
KOAST	ZERDAT	=-1, SHORT COAST. =+1, LONG-COAST
KOUNT	COM	A CODE WHICH SPECIFIES WHETHER PRINT-OUT IS TO OCCUR AFTER THIS TIME INCREMENT
KPRINT	XXXL	CORRELATION MATRIX PRINT CODE
KSICA	TARINT	STOPPING CONDITION INDICATOR IN SUBROUTINE TRJTRY =1, STOP ON TIME =2, STOP AT SPHERE-OF-INFLUENCE =3, STOP AT CLOSEST APPROACH
KTAR(6,10)	TARINT	CODES OF TARGET PARAMETERS (UP TO 6) FOR EACH TARGETING EVENT OR ORBIT INSERTION OPTION FOR EACH INSERTION EVENT
KTIM(10)	TARINT	EPOCH TO WHICH GUIDANCE EVENT TIMES ARE REFERENCED -0 EVENT NOT PROCESSED =1 INITIAL TIME =2 SOI TIME =3 CA TIME =4 CALENDAR DATE
KTYP(10)	TARINT	TYPE OF GUIDANCE EVENT FOR EACH EVENT =-1 TERMINATION EVENT =1 TARGETING EVENT =2 RETARGETING EVENT =3 ORBIT INSERTION EVENT
KMXQ(10)	TARINT	COMPUTE/EXECUTE MODES FOR EACH GUIDANCE EVENT =1 COMPUTE VELOCITY CORRECTION ONLY =2 EXECUTE VELOCITY CORRECTION ONLY =3 COMPUTE AND IMMEDIATELY EXECUTE CORRECTION =4 COMPUTE BUT EXECUTE CORRECTION LATER

KUR	TARINT	FLAG INDICATING WHICH EVENT IS THE CURRENT EVENT
KWIT	TARINT	TERMINATION INDICATOR =0, CONTINUE RUN =1, TERMINATE RUN
LEV	TARINT	CURRENT LEVEL IN CURRENT TARGETING EVENT
LEVELS	TARINT	NUMBER OF ACCURACY LEVELS FOR CURRENT EVENT
LINCT	COM	LINE COUNT (NOMNAL ONLY)
LINPGE	COM	LINES PER PAGE (NOMNAL ONLY)
LKLP(10)	TARVAR	ARRAY OF LAUNCH PLANETS
LKTAR(6,10)	TARVAR	ARRAY DEFINING TARGET PARAMETERS
LKTP(10)	TARVAR	ARRAY OF TARGET PLANETS
LLVLS(10)	TARVAR	NUMBER OF INTEGRATION ACCURACY LEVELS USED
LNPAR(10)	TARVAR	NUMBER OF TARGET PARAMETERS DESIRED
LTARG	ZERDAT	= 0, INTERPLANETARY TARGETING. =1, LUNAR TARGETING
LVLS	TARINT	NUMBER OF ACCURACY LEVELS TO BE USED ON EACH TARGETING EVENT
MAT(10)	TARINT	TARGETING MATRIX COMPUTATION CODE FOR EACH TARGETING EVENT =1 COMPUTE TARGETING MATRIX ONLY AT FIRST LEVEL -2 COMPUTE TARGETING MATRIX AT EACH STEP
MATX	TARINT	MATRIX COMPUTATION CODE FOR CURRENT TARGETING EVENT (SEE DEFN OF MAT)
MAXB(10)	TARINT	THE NUMBER OF BAD STEPS ALLOWED DURING ANY TARGETING EVENT
MAXBAD	TARINT	MAXIMUM NUMBER OF BAD ITERATIONS FOR CURRENT EVENT
MCNTR	MEAS	NUMBER OF MEASUREMENTS HAVING OCCURRED
MCODE(1000)	MEAS	ARRAY OF MEASUREMENT CODES

MDL(10)	TARINT	EXECUTION MODELS FOR EACH GUIDANCE EVENT =1 IMPULSIVE =2 PULSING ARC
MNCN(12)	CONST	MEASUREMENT NOISE CONSTANTS
MNNAME(12,3)	NAME	MEASUREMENT NAME
MONTH(12)	PRT	NAMES OF MONTHS
NAE	EVENT	ADAPTIVE FILTERING EVENTS HAVING OCCURRED.
NAF6(20)	EVENT	ARRAY OF ADAPTIVE FILTERING EVENT CODES. NON-FUNCTIONAL IN EXISTING PROGRAM
NAFC	OVER1	ADAPTIVE FILTER FLAG
NB(11)	VM	CODES OF PLANETS
NB1(11)	SIM2	ARRAY OF PLANET CODES IN ACTUAL TRAJECTORY
NBOD	VM	NUMBER OF BODIES USED IN VIRTUAL MASS PROGRAM
NBOD1	SIM2	NUMBER OF BODIES IN ACTUAL TRAJECTORY
NBODY	COM	EQUAL TO 4*NBODYI-3
NBODYI	COM	NUMBER OF BODIES CONSIDERED IN VIRTUAL MASS TRAJECTORY
NCPR	TARINT	NUMBER OF INTEGRATION INCREMENTS BETWEEN PRINT-OUTS OF NOMINAL TRAJECTORY
NDACC	TRAJCD	NUMERICAL DIFFERENCING ACCURACY CODE
NDIM1	STVEC	DIMENSION OF SOLVE-FOR PARAMETER STATE
NDIM2	STVEC	DIMENSION OF DYNAMIC CONSIDER STATE
NDIM3	STVEC	DIMENSION OF MEASUREMENT CONSIDER PARAMETER STATE
NEV	EVENT	NUMBER OF EVENTS
NEV1	EVENT	TOTAL NUMBER OF EIGENVECTOR EVENTS
NEV2	EVENT	TOTAL NUMBER OF PREDICTION EVENTS

NEV3	EVENT	TOTAL NUMBER OF GUIDANCE EVENTS
NEV4	EVENT	TOTAL NUMBER OF -COMCON- EVENTS
NEV5	EVENT	TOTAL NUMBER OF QUASI-LINEAR FILTERING EVENTS
NEV6	EVENT	TOTAL NUMBER OF ADAPTIVE FILTERING EVENTS. NON-FUNCTIONAL IN EXISTING PROGRAM
NEV7	EVENT	ORBIT INSERTION VARIABLES. NON-FUNCTIONAL IN IN EXISTING PROGRAM
NEV8	EVENT	ORBIT INSERTION VARIABLE. NON-FUNCTIONAL IN EXISTING PROGRAMS
NEV9	EVENT	ORBIT INSERTION VARIABLES. NON-FUNCTIONAL IN IN EXISTING PROGRAM
NEV10	EVENT	ORBIT INSERTION VARIABLES. NON-FUNCTIONAL IN IN EXISTING PROGRAM
NEV11	EVENT	ORBIT INSERTION VARIABLES. NON-FUNCTIONAL IN IN EXISTING PROGRAM
NGE	EVENT	NUMBER OF GUIDANCE EVENTS HAVING OCCURRED
NINETY	DPNUM	THE NUMBER NINETY (90) TO NINE SIGNIFICANT FIGURES. TARGETING MODE ONLY
NITS	TARINT	ALLOWABLE NUMBER OF ITERATIONS FOR CURRENT EVENT
NLP	VM	CODE OF LAUNCH PLANET
NMN	MEAS	TOTAL NUMBER OF MEASUREMENTS
NO(11)	BLK	AN ARRAY OF PLANET CODES BEING USED TO GENERATE THE VIRTUAL MASS TRAJECTORY
NOGYD	TARINT	TOTAL NUMBER OF GUIDANCE EVENTS
NOIT(10)	TARINT	THE NUMBER OF TOTAL ITERATIONS ALLOWED AT THE FIRST AND LAST LEVELS OF TARGETING EVENTS FOR EACH GUIDANCE EVENT
NOPAR	TARINT	NUMBER OF TARGET PARAMETERS FOR CURRENT EVENT

NOPHAS	TARINT	NUMBER OF TARGETING PHASES FOR CURRENT EVENT
NOSOI	TARINT	OUTER TARGETING FLAG =0 NORMAL TARGETING =1 OUTER TARGETING
NPE	EVENT	NUMBER OF PREDICTION EVENTS HAVING OCCURRED
NPUL	PULS	NUMBER OF PULSES IN PULSING ARC
NQE	EVENT	QUASI-LINEAR FILTERING EVENTS HAVING OCCURRED
NR	OVER1	NUMBER OF ROWS IN THE OBSERVATION MATRIX
NST	CONST	NUMBER OF STATIONS TO BE USED (MAXIMUM 3)
NTMC	TRAJCD	NOMINAL TRAJECTORY CODE
NTP	VM	CODE OF TARGET PLANET
OMEGA	CONST	EARTH-S ROTATION RATE
ONE	DPNUM	THE NUMBER ONE (1) TO NINE SIGNIFICANT FIGURES
OTAR(3)	LUNART	DESIRED VALUES OF SMA,RCA, AND INC
P(6,6)	STM	POSITION/VELOCITY COVARIANCE
PB(6,6)	STM	POSITION/VELOCITY COVARIANCE AT INITIAL TIME
P7	EVENT	ORBIT INSERTION VARIABLES. NON-FUNCTIONAL IN EXISTING PROGRAM
PCON(3)	LUNART	PERTURBATIONS IN CONTROLS(ALPHA,DELTA, THETA)
PERP7	EVENT	ORBIT INSERTION VARIABLES. NON-FUNCTIONAL IN EXISTING PROGRAM
PERV(10)	TAREAL	PERTURBATIONS SIZE FOR VELOCITY COMPONENTS IN CONSTRUCTING SENSITIVITY MATRICES IN TARGETING EVENTS
PG(6,6)	GUI	POSITION/VELOCITY CONTROL COVARIANCE

PHI(3,3)	TAREAL	TARGETING MATRIX. REQ-D ONLY IF ISTART=1,2
PHI(6,6)	STM	POSITION/VELOCITY STATE TRANSITION MATRIX
PHI2(3,3)	BAIM	INVERSE OF VARIATION MATRIX PARTITION
PHILS	ZERDAT	LATITUDE OF LAUNCH SITE
PI	COM	THE VALUE OF THE MATHEMATICAL CONSTANT PI
PLANET(11)	PRT	NAMES OF PLANETS
PMASS(11)	BLK	GRAVITATIONAL CONSTANTS OF PLANETS IN A.U.**3/DAY**2
POI	PBLK	PROBABILITY OF IMPACT
PP(6,6)	STM	POSITION/VELOCITY COVARIANCE MATRIX PRIOR TO PROCESSING A MEASUREMENT
PROBI	BAIM	ALLOWABLE PROBABILITY OF IMPACT
PS(24,24)	STM	SOLVE-FOR PARAMETER COVARIANCE
PSB(24,24)	STM	SOLVE-FOR PARAMETER COVARIANCE AT INITIAL TIME
PSG(24,24)	GJI	SOLVE-FOR PARAMETER CONTROL COVARIANCE
PSI1	ZERDAT	ANGLE OF FIRST BURN
PSI2	ZERDAT	ANGLE OF SECOND BURN
PSP(24,24)	STM	SOLVE-FOR PARAMETER COVARIANCE MATRIX PRIOR TO PROCESSING A MEASUREMENT
PSTAR	PBLK	NOMINAL PROBABILITY DENSITY FUNCTION EVALUATED AT TARGET PLANET CENTER
PULMAG	PULS	THRUST MAGNITUDE OF PULSING ENGIEN
PULMAS	PULS	NOMINAL MASS OF SPACECRAFT DURING PULSING ARC
PULT	PULS	TOTAL TIME INTERVAL OF PULSING ARC

Q(6,6)	STM	DYNAMIC NOISE COVARIANCE MATRIX
QK(6,6)	EXE	EFFECTIVE EXECUTION ERROR COVARIANCE MATRIX
R(4,4)	STM	MEASUREMENT NOISE COVARIANCE MATRIX
RAP	ZOUT	RIGHT ASCENSION OF VHP
RAD	COM	THE NUMBER OF DEGREES PER RADIAN
RADIUS(11)	BLK	THE RADIUS OF A GIVEN PLANET IN A.U.
RC(6)	VM	STATE AT CLOSEST APPROACH
RCA	VM	MAGNITUDE OF CLOSEST APPROACH POSITION VECTOR
RCA	LUNART	RADIUS OF CLOSEST APPROACH TO MOON (DESIRED)
RCA1(6)	TRJ	STATE AT CLOSEST APPROACH ON ORIGINAL NOMINAL
RCA2(6)	TRJ	STATE AT CLOSEST APPROACH ON MOST RECENT NOMINAL
RCA3(6)	TRJ	STATE AT CLOSEST APPROACH ON ACTUAL TRAJECTORY
RE(6)	VM	POSITION AND VELOCITY OF EARTH
RES(4)	SIM1	RESIDUAL
RF(6)	OVER	FINAL TARGETED NOMINAL STATE VECTOR
RF(6)	OVERZ	FINAL TARGETED STATE VECTOR
RF1(6)	OVER	FINAL MOST RECENT NOMINAL STATE VECTOR
RF1(6)	OVERZ	FINAL MOST RECENT NOMINAL STATE VECTOR
RI(6)	OVER1	INITIAL TARGETED NOMINAL STATE VECTOR
RI(6)	OVERR	STATE VECTOR AT EVENT TIME
RI(6)	LUNART	GEOCENTRIC STATE OF S/C AT LUNAR SOI
RI1	OVER1	INITIAL MOST RECENT NOMINAL STATE VECTOR

RIN(6)	TAREAL	CURRENT STATE VECTOR AT I-TH EVENT
RK(2,3)	PULS	POSITION VECTORS OF LAUNCH AND TARGET PLANETS AT IMPULSIVE TIME(MIDPOINT OF PULSING ARC)
RMASS(11)	BLK	THE RELATIVE GRAVITATIONAL CONSTANT OF A STATED PLANET WITH RESPECT TO THE SUN
RME(6)	LUNART	GEOCENTRIC STATE OF CENTER OF MOON IN ECLIPTIC COORDINATES AT TSI
RMQ(6)	LUNART	GEOCENTRIC STATE OF CENTER OF MOON AT TSI IN EQUATORIAL COORDINATES
RP	ZERDAT	PARKING ORBIT RADIUS
RPE	LUNART	RADIUS OF EARTH PARKING ORBIT
RPRAT	ZERDAT	PARKING ORBIT INVERSE RATE
RRF(3)	TAREAL	SPACECRAFT POSITION AT END OF INTEGRATION, USED IN BROKEN PLANE TARGETING
RSI(3)	VM	POSITION AT SPHERE OF INFLUENCE
RSI(6)	LUNART	SELENOCENTRIC STATE OF S/C AT LUNAR SOI
RSOI1(3)	TRJ	POSITION AT SPHERE OF INFLUENCE ON ORIGINAL NOMINAL
RSOI2(3)	TRJ	POSITION AT SPHERE OF INFLUENCE ON MOST RECENT NOMINAL
RSOI3(3)	TRJ	POSITION AT SPHERE OF INFLUENCE ON ACTUAL TRAJECTORY
RTP(6)	VM	POSITION AND VELOCITY OF TARGET PLANET
RVS(6)	VM	POSITION OF VEHICLE RELATIVE TO VIRTUAL MASS
S(24,4)	STM	KALMAN GAIN CONSTANT FOR SOLVE-FOR PARAMETERS
SAL(3)	CONST	ALTITUDES OF STATIONS
SIGALP	EVENT	VARIANCE OF ERROR IN POINTING ANGLE 1

SIGBET	EVENT	VARIANCE OF ERROR IN POINTING ANGLE 2
SIGMAL	ZERDAT	NOMINAL LAUNCH AZIMUTH
SIGPRO	EVENT	VARIANCE OF PROPORTIONALITY ERROR
SIGRES	EVENT	VARIANCE OF RESOLUTION ERROR
SLAT(3)	CONST	LATITUDES OF STATIONS
SLB(9)	SIMCNT	BIASES IN STATION LOCATION CONSTANTS
SLON(3)	CONST	LONGITUDES OF STATIONS
SMA	LUNART	SIMI-MAJOR AXIS OF LUNAR HYPERBOLA (DESIRED)
SMJR(18)	BLK	CONSTANTS USED TO CALCULATE THE SEMI-MAJOR AXES OF THE PLANETS
SPHERE(11)	BLK	THE SPHERES OF INFLUENCE OF THE PLANETS IN A.U.
SPHFAC(10)	TAREAL	REDUCTION FACTORS FOR TARGET PLANET SPHERE OF INFLUENCE FOR EACH EVENT
SSS(3)	VM	DIRECTION COSINES VECTOR OF SPACECRAFT SPIN AXIS
ST(50)	BLK	CONSTANTS USED TO CALCULATE THE ORBITAL ELEMENTS OF THE LAST FOUR PLANETS (SEE LARGE ARRAY DEFNS IN SECTION 5.3)
T	BLK	TRAJECTORY TIME IN DAYS
T1(20)	TMW2	EIGENVECTOR EVENT TIMES
T2(20)	TMW2	PREDICTION EVENT STARTING TIMES
T3(10)	BAIM	ARRAY OF GUIDANCE EVENT TIMES
T4(20)	TMW2	CONIC COMPUTATION EVENT TIMES
T5(20)	TMW2	QUASI-LINEAR EVENT TIMES

T6(20)	THW2	NOT USED
T7	THW2	NOT USED
TACA	VM	TRAJECTORY SEMIMAJOR AXIS WITH RESPECT TO TARGET BODY AT CLOSEST APPROACH TO TARGET BODY
TAR(6,10)	TAREAL	DESIRED VALUES OF TARGET PARAMETERS (UP TO 6 AVAILABLE) FOR EACH GUIDANCE EVENT
TAU7	EVENT	ORBIT INSERTION VARIABLES. NON-FUNCTIONAL IN IN EXISTING PROGRAM
TCA	LUNART	J.D. OF TIME AT LUNAR CLOSEST APPROACH (DESIRED)
TCA1	TRJ	TIME AT CLOSEST APPROACH OF ORIGINAL NOMINAL
TCA2	TRJ	TIME AT CLOSEST APPROACH OF MOST RECENT NOMINAL
TCA3	TRJ	TIME AT CLOSEST APPROACH OF ACTUAL TRAJECTORY
TEN	DPNUM	THE NUMBER TEN (10) TO NINE SIGNIFICANT FIGURES. TARGETING MODE ONLY
TEV(50)	EVENT	TIMES OF EVENTS
TEVN	OVER1	TIME OF CURRENT EVENT
TEVN	OVERR	EVENT TIME
TF	ZERDAT	NOT USED
TG	GUI	TRAJECTORY TIME AT MOST RECENT GUIDANCE EVENT
TGT3(10)	TARVAR	DESIRED TARGET TIMES REFERENCED TO INITIAL TRAJECTORY TIME
THEDOT	ZERDAT	ROTATION RATE OF LAUNCH PLANET
THELS	ZERDAT	LONGITUDE OF LAUNCH SITE

THREE	OPNUM	THE NUMBER THREE (3) TO NINE SIGNIFICANT FIGURES
TI	ZERDAT	NOT USED
TIMG(10)	TAREAL	TIMES OF EACH GUIDANCE EVENT REFERENCED TO EPOCH-INITIAL TIME, SOI TIME, OR CA TIME
TIM1	ZERDAT	TIME OF FIRST BURN
TIM2	ZERDAT	TIME OF SECOND BURN
TIMINT	VM	TOTAL TIME USED
TIMS	TAREAL	INTERNAL CLOCK TIME AT START OF COMPUTER RUN
TIN	TAREAL	JULIAN DATE AT INJECTION
TINJ	BAIM	INJECTION TIME
TM	VM	TIME UNITS PER DAY
TMN(1050)	MEAS	TIMES OF MEASUREMENTS
TMPR(3)	BAIM	MOST RECENT TARGET STATE
TNOMB(3)	BAIM	NOMINAL B-PLANE TARGET STATE
TNOMC(7)	BAIM	NOMINAL CLOSEST APPROACH TARGET STATE
TMU	TAREAL	GRAVITATIONAL CONSTANT OF TARGET PLANET
TMU	LUNART	GRAVITATIONAL CONSTANT OF MOON(KM3/SEC2)
TOL(6,10)	TAREAL	ALLOWABLE TOLERANCES OF TARGET PARAMETERS FOR EACH GUIDANCE EVENT
TRTM	TAREAL	TRAJECTORY TIME (DAYS) REF. TO INJECTION
TOL(6,10)	TAREAL	TOLERANCES OF TARGET PARAMETERS FOR EACH TARGETING EVENT
TOLR(6)	TAREAL	NOT USED IN CURRENT TARGET VERSION

TPT2(20)	EVENT	ARRAY OF TIMES TO WHICH A PREDICTION IS MADE
TRTM	TAREAL	TRAJECTORY TIME ON NOMINAL TRAJECTORY
TRTM1	TIME	INITIAL TRAJECTORY TIME
TRTMB	TIME	TRAJECTORY TIME AT BEGINNING OF TRAJECTORY
TSI	LUNART	PROJECTED J.D. AT SOI INERTSECTION
TSOI1	TRJ	TIME AT SPHERE OF INFLUCNCE OF ORIGINAL NOMINAL
TSOI2	TRJ	TIME AT SPHERE OF INFLUENCE OF MOST RECENT NOMINAL
TSOI3	TRJ	TIME AT SPHERE OF INFLUENCE OF ACTUAL TRAJECTORY
TSPH	LUNART	RADIUS OF LUNAR SOI (KM)
TTIM1	SIMCNT	FIRST TIME USED FOR UNMODELLED ACCELERATION
TTIM2	SIMCNT	SECOND TIME USED FOR UNMODELLED ACCELERATION
TTOL(3)	LUNART	ALLOWABLE TOLERANCES IN SMA,B.T,B.R
TWO	DPNUM	THE NUMBER TWO (2) TO NINE SIGNIFICANT FIGURES
TWOPI	DPNUM	THE MATHEMATICAL CONSTANT 2.*PI
TXU(6,8)	STM	STATE TRANSITION MATRIX PARTITION ASSOCIATED WITH DYNAMIC CONSIDER PARAMETERS
TXXS(6,24)	STM	STATE TRANSITION MATRIX PARTITION ASSOCIATED WITH SOLVE-FOR PARAMETERS
U0(3,8)	STM	DYNAMIC CONSIDER PARAMETER COVARIANCE MATRIX
UNIVT	TIME	UNIVERSAL TIME
UNMAC(3,3)	SIMCNT	UNMODELLED ACCELERATION
UST(3)	CONST2	DIRECTION COSINES OF 3 REFERENCE STARS

V(16,7)	COM	AN ARRAY WHICH STORES PERTINANT VECTORS USED IN THE CALCULATION OF THE VIRTUAL MASS TRAJECTORY (SEE LARGE ARRAY DEFNS IN SECTION 5.3)
VO(15,15)	STM	MEASUREMENT CONSIDER PARAMETER COVARIANCE MATRIX
VHPM	ZOUT	MAGNITUDE OF HYPERBOLIC EXCESS VELOCITY AT TARGET BODY (VHP VECTOR)
VINF	BAIM	HYPERBOLIC EXCESS VOLOCITY
VK(2,3)	PULS	VELOCITY VECTORS OF LAUNCH AND TARGET PLANETS AT IMPULSIVE TIME (MIDPOINT OF PULSING ARC)
VMU	VM	GRAVITATIONAL CONSTANT OF VIRTUAL MASS
VSI(3)	VM	VELOCITY AT SPHERE OF INFLUENCE
VSOI1(3)	TRJ	VELOCITY AT SPHERE OF INFLUENCE ON ORIGINAL NOMINAL
VSOI2(3)	TRJ	VELOCITY AT SPHERE OF INFLUENCE ON MOST RECENT NOMINAL
VSOI3(3)	TRJ	VELOCITY AT SPHERE OF INFLUENCE ON ACTUAL TRAJECTORY
VST(3)	CONST2	DIRECTION COSINES OF 3 REFERENCE STARS
W(17)	SIM1	ACTUAL DYNAMIC NOISE
WST(3)	CONST2	DIRECTION COSINES OF 3 REFERENCE STARS
XAC(5,10)	TARVAR	ACCURACY LEVELS EMPLOYED IN TARGETING
XB(6)	STVEC	BEGINNING ORIGINAL NOMINAL VEHICLE STATE VECTOR
XBDT	SAVVAL	ORIGINAL VALUE OF B.T IN N-L GUIDANCE
XBDR	SAVVAL	ORIGINAL VALUE OF B.R IN N-L GUIDANCE
XDC	SAVVAL	ORIGINAL VALUE OF DC IN N-L GUIDANCE
XDELV(3,10)	TARVAR	NONLINEAR VELOCITY CORRECTION

XDSI	SAVVAL	ORIGINAL VALUE OF TSI IN N-L GUIDANCE
XDVMAX(10)	TARVAR	MAXIMUM ALLOWABLE VELOCITY CORRECTION
XF(6)	STVEC	FINAL VEHICLE STATE VECTOR OF ORIGINAL NOMINAL
XF1(6)	SIM1	FINAL STATE VECTOR OF MOST RECENT NOMINAL TRAJECTORY
XFAC(10)	TARVAR	SPHERE OF INFLUENCE FACTORS
XG(6)	GUI	STATE VECTOR AT TIME OF LAST GUIDANCE EVENT
XI(6)	STVEC	INITIAL VEHICLE STATE VECTOR OF ORIGINAL NOMINAL
XI1(6)	SIM1	INITIAL STATE VECTOR OF MOST RECENT NOMINAL
XIN(6)	OVERX	STATE VECTOR TRANSFERRED TO NONLIN
XLAB(6)	XXXL	VEHICLE POSITION/VELOCITY VECTOR COMPONENT NAMES
XLAM(2,2)	PBLK	PROJECTION OF TARGET CONDITION COVARIANCE MATRIX INTO THE IMPACT PLANE
XLAMI(2,2)	PBLK	INVERSE OF XLAM(2,2)
XMUS(2)	PBLK	NOMINAL IMPACT PLANE TARGET STATE
XNM(24)	XXXL	AUGUMENTATION PARAMETER LABELS
XP(6)	BLK	THE POSITION AND VELOCITY OF A PLANET IN HELIOCENTRIC ECLIPTIC COORDINATES
XPERV(10)	TARVAR	VELOCITY PERTURBATION USED TO COMPUTE TARGETING MATRIX
XRC(6)	SAVVAL	ORIGINAL VALUE OF RC IN N-L GUIDANCE
XRSI(3)	SAVVAL	ORIGINAL VALUE OF RSI IN N-L GUIDANCE

XSL(24)	XXXL	SOLVE-FOR PARAMETER LABELS
XTAR(6,10)	TARVAR	DESIRED TARGET VALUES ·
XTOL(6,10)	TARVAR	TOLERANCES ON TARGET PARAMETERS
XU(8)	XXXL	DYNAMIC CONSIDER PARAMETER LABELS
XV(15)	XXXL	MEASUREMENT CONSIDER PARAMETER LABELS
XVSI(3)	SAVVAL	ORIGINAL VALUE OF VSI IN N-L GUIDANCE
XXIN(6)	EXE	STATE VECTOR TRANSFERRED TO EXCUT OR EXCUTS
Z(17)	SIM1	ACTUAL STATE VECTOR
ZDAT(6)	ZERDAT	ZERO ITERATE VECTOR IF IZERO=0 ZDAT(1-6)= INITIAL STATE =2,3,4, ZDAT(1-3)= INITIAL POSITION ZDAT(4-6)= FINAL POSITION
ZERO	DPNUM	THE NUMBER ZERO (0) TO NINE SIGNIFICANT FIGURES
ZI(17)	SIM1	INITIAL ACTUAL STATE VECTOR
ZF(6)	SIM1	FINAL ACTUAL STATE VECTOR AFTER ADDING THE EFFECT OF UNMODELED ACCELERATION

5.3 Large Array Definitions

In this section large arrays appearing in COMMON will be displayed. The arrays depicted are frequently referenced in trajectory propagation subroutines in STEAP; hence the programmer studying such subroutines will find the following tables extremely useful.

Tables 5.1 to 5.5 describe arrays containing planetary ephemeris constants. The values actually stored in these arrays may be found in the documentation for BLOCK DATA. Tables 5.6 through 5.8 contain variables used in the virtual mass propagation procedure. Discussions of these variables may be found in VMP, EPHEM, ORB, and similar routines.

Constant	i	Ω	$\tilde{\omega}$	e	M	a	ω	E	a_0	a_1
Mercury	1	2	3	4	5	6	7	8	1	2
Venus	9	10	11	12	13	14	15	16	3	4
Earth	17	18	19	20	21	22	23	24	5	6
Mars	25	26	27	28	29	30	31	32	7	8
Jupiter	33	34	35	36	37	38	39	40	9	10
Saturn	41	42	43	44	45	46	47	48	11	12
Uranus	49	50	51	52	53	54	55	56	13	14
Neptune	57	58	59	60	61	62	63	64	15	16
Pluto	65	66	67	68	69	70	71	72	17	18
Moon	73	74	75	76	77	78	79	80		

Table 5.1 ELMNT Array -- Conic Elements Table 5.2 SMJR Array

Constant	i_0	i_1	i_2	i_3	Ω_0	Ω_1	Ω_2	Ω_3	$\tilde{\omega}_0$	$\tilde{\omega}_1$	$\tilde{\omega}_2$	$\tilde{\omega}_3$	e_0	e_1	e_2	e_3	M_0	M_1	M_2	M_3
Mercury	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16	17	18	19	20
Venus	21	22	23	24	25	26	27	28	29	30	31	32	33	34	35	36	37	38	39	40
Earth	41	42	43	44	45	46	47	48	49	50	51	52	53	54	55	56	57	58	59	60
Mars	61	62	63	64	65	66	67	68	69	70	71	72	73	74	75	76	77	78	79	80

Table 5.3 CN Array -- Inner Planet Constants

Constant	i_0	i_1	Ω_0	Ω_1	$\tilde{\omega}_0$	$\tilde{\omega}_1$	e_0	e_1	M_0	M_1
Jupiter	1	2	3	4	5	6	7	8	9	10
Saturn	11	12	13	14	15	16	17	18	19	20
Uranus	21	22	23	24	25	26	27	28	29	30
Neptune	31	32	33	34	35	36	37	38	39	40
Pluto	41	42	43	44	45	46	47	48	49	50

Table 5.4 ST Array -- Outer Planet Constants

Constant	0	1	2	3	0	1	2	3	L_0	L_1	L_2	L_3	i	e	a
Moon	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15

Table 5.5 EMN Array -- Lunar Constants

F-Array

x_1	y_1	z_1	r_1
$\dot{x}_1$	$\dot{y}_1$	$\dot{z}_1$	v_1
x_{S1}	y_{S1}	z_{S1}	r_{S1}
$\dot{x}_{S1}$	$\dot{y}_{S1}$	$\dot{z}_{S1}$	v_{S1}
x_2	y_2	z_2	r_2
$\dot{x}_2$	$\dot{y}_2$	$\dot{z}_2$	v_2
x_{S2}	y_{S2}	z_{S2}	r_{S2}
$\dot{x}_{S2}$	$\dot{y}_{S2}$	$\dot{z}_{S2}$	v_{S2}
⋮	⋮	⋮	⋮

Note:

Subscript i indicates component is i-th body referenced to inertial coordinate system

Subscript si indicates component is spacecraft referenced to i-th body

Table 5.6 F-Array -- Ephemeris Data

TRG(1)	cos E	TRG(5)	cos i	TRG(9)	cos ω	TRG(13)	cos(ω+ f)
TRG(2)	sin E	TRG(6)	sin i	TRG(10)	sin ω	TRG(14)	sin(ω+ f)
TRG(3)	cos f	TRG(7)	cos Ω	TRG(11)	cos ω		
TRG(4)	sin f	TRG(8)	sin Ω	TRG(12)	sin ω		

Table 5.7 TRG Array -- Trigonometric Functions

I \ J	1	2	3	4	5	6	7
1	$(t_e)_{dim}, t_B$	$(x_{s_e})_{dim}, x_{s_B}$	$(y_{s_e})_{dim}, y_{s_B}$	$(z_{s_e})_{dim}, z_{s_B}$	ω (deg/t), ω (rad/t)	D	$\mu, \neg\mu$
2	t_e	x_{s_e}	y_{s_e}	z_{s_e}	$(t_P)_{dim}, t_P$	$(r_{1s_P})_{dim}, r_{1s_P}$	$(r_{2s_P})_{dim}, r_{2s_P}$
3	$(t_{eph_e})_{dim}, t_{eph_B}$	$(\dot{x}_{s_e})_{dim}, \dot{x}_{s_B}$	$(\dot{y}_{s_e})_{dim}, \dot{y}_{s_B}$	$(\dot{z}_{s_e})_{dim}, \dot{z}_{s_B}$	$(\Delta t_P)_{dim}, \Delta t_P$	C_2	
4	t_{eph_e}	$\dot{x}_{s_e}$	$\dot{y}_{s_e}$	$\dot{z}_{s_e}$	$\frac{\Delta r}{r}$		ωD (velocity)
5	$(\mu_{v_e})_{dim}, \mu_{v_B}$	$(x_{v_e})_{dim}, x_{v_B}$	$(y_{v_e})_{dim}, y_{v_B}$	$(z_{v_e})_{dim}, z_{v_B}$	ωD^2 (area rate)	$\omega^2 D^3$ (velocity)^2	$1 - \mu$
6	μ_{v_e}	M_x, x_{v_e}	M_y, y_{v_e}	M_z, z_{v_e}	$\omega^2 D^3$ (mass)	$\omega^3 D^3$ (mass rate)	$r, \Delta v$
7	$(\dot{\mu}_{v_e})_{dim}, \dot{\mu}_{v_B}$	$(\dot{x}_{v_e})_{dim}, \dot{x}_{v_B}$	$(\dot{y}_{v_e})_{dim}, \dot{y}_{v_B}$	$(\dot{z}_{v_e})_{dim}, \dot{z}_{v_B}$	Δt_k	Δt	$\mu_{v_{average}}$
8	$\dot{\mu}_{v_e}$	$M_x, \dot{x}_{v_e}$	$M_y, \dot{y}_{v_e}$	$M_z, \dot{z}_{v_e}$	$(\Delta t)^2$	$\dot{\mu}_{v_{average}}$	$D, \neg M$
9	r_{vs_B}	x_{vs_B}	y_{vs_B}	z_{vs_B}	$x_{vs_B}, (\sigma_{vs_e})_x$	$y_{vs_B}, (\sigma_{vs_e})_y$	$z_{vs_B}, (\sigma_{vs_e})_z$
10	r_{vs_e}	x_{vs_e}	y_{vs_e}	z_{vs_e}	$x_{v_{avg}}, \ddot{x}_{v_{avg}}$	$y_{v_{avg}}, \ddot{x}_{v_{avg}}$	$\dot{z}_{v_{avg}}, z_{v_{avg}}$
11	V_{vs_B}	$\dot{x}_{vs_B}$	$\dot{y}_{vs_B}$	$\dot{z}_{vs_B}$	$\dot{x}_{vs_B}, a_x$	$\dot{y}_{vs_B}, a_y$	$\dot{z}_{vs_B}, a_z$
12		$\dot{x}_{vs_e}, e_x + \frac{x_{vs_e}}{r_{vs_e}}$	$y_{vs_e}, e_y + \frac{y_{vs_e}}{r_{vs_e}}$	$\dot{z}_{vs_e}, e_z \frac{z_{vs_e}}{r_{vs_e}}$	$M_s, e_x + \frac{x_{vs_e}}{r_{vs_e}}$	$M_s, e_y + \frac{y_{vs_e}}{r_{vs_e}}$	$e_z + \frac{z_{vs_e}}{r_{vs_e}}$
13			t_p	Δt_{MA}	$1. - e_e^2$	$(\lvert 1. - e_e^2 \rvert)^{1/2}$	$\frac{h^2}{\mu_{v_{avg}}}$
14	e_e	e_{x_e}	e_{y_e}	e_{z_e}	e_e^2, e_{x_e}	cos (t_e), e_{y_e}	sin (t_e), e_{z_e}
15	$(h)_{dim}$	$(h_x)_{dim}$	$(h_y)_{dim}$	$(h_z)_{dim}$	b_B, X_B	E_B	$a \cdot r_{vs_B}$, or $\vec{r}_{vs_B} \cdot \vec{r}_{vs_B}$
16	h	h_x	h_y	h_z	h_x, h_e^2, b_e, X_e	h_y, E_e	$h_z, a - r_{vs_e}$; or $\vec{r}_{vs_e} \cdot \vec{r}_{vs_B}$

Table 5.8 V-Array -- Virtual Mass Propagation Variables

6. INDIVIDUAL SUBROUTINE DOCUMENTATION

This chapter contains the individual documentation for all the subroutines in the STEAP II series. The following information is given for each subroutine.

1. Purpose: The tasks performed by the subroutine.

2. Calling Sequence: The statement by which the subroutine is called.

3. Arguments: The arguments in the calling sequence, their definition, and identification as input, output, or both.

4. Subroutines Supported: A list of subroutines calling the subroutine being documented.

5. Subroutines Required: A list of subroutines called by the subroutines being documented.

6. Local Symbols: The internal (non-common) variables used in the subroutine and their definitions.

7. Common Computed/Used: A list of variables appearing in common blocks which are both computed and used (see Chapter 3 for definitions).

8. Common Computed: A list of common variables which are set in the program.

9. Common Used: A list of common variables only used by the subroutine.

10. Analysis: The detailed mathematical analysis on which the subroutine is based (if applicable).

11. Flowchart: A flowchart of the operation of the program (if required).

The reader is referred to Chapter 4 for an index of all subroutines of STEAP II (Tables 4.1 and 4.2) and for the calling hierarchies of the basic subprograms of STEAP II (Figures 4.1 to 4.4).

SUBROUTINE BATCON

PURPOSE: BATCON IS A CONIC PROPAGATOR USING THE BATTIN UNIVERSAL VARIABLE FORMULATION.

CALLING SEQUENCE: CALL BATCON(GM,RV,VV,DT,DV,SV)

ARGUMENTS:

GM	I	GRAVITATIONAL CONSTANT
RV(3)	I	INITIAL POSITION VECTOR
VV(3)	I	INITIAL VELOCITY VECTOR
DT	I	TIME INTERVAL OF PROPAGATION
DV(3)	O	FINAL POSITION VECTOR
SV(3)	O	FINAL VELOCITY VECTOR

SUBROUTINES SUPPORTED: PERHEL

SUBROUTINES REQUIRED: NONE

LOCAL SYMBOLS:

ALP	BATTIN ALPHA VARIABLE
ASQ	SQRT OF ALP
CONT	INTERMEDIATE VARIABLE
CON	NORMALIZED TIME
DELX	CORRECTION TO CURRENT VALUE OF X
EX	E RAISED TO THE ASQ*X POWER
FT	F SERIES FOR VELOCITY EVALUATED AT EPOCH
F	F SERIES FOR POSITION EVALUATED AT EPOCH
GMSQ	SQRT OF GRAVITIONAL CONSTANT
GT	G SERIES FOR VELOCITY EVALUTED AT EPOCH
G	G-SERIES COEFFICIENT FOR POSITION
I	KEPLER-LIKE EQUATION ITERATION COUNTER
RI	NAME FOR COMPONENTS OF INITIAL POSITION
R0	MAGNITUDE OF INITIAL POSITION
R2	SQUARE OF R0

R	ITERATE VALUE OF RADIUS
SIG	INTERMEDIATE VARIABLE
U0	BATTIN TRANSCENDENTAL FUNCTION
U1	BATTIN TRANSCENDENTAL FUNCTION
U2	BATTIN TRANSCENDENTAL FUNCTION
U3	BATTIN TRANSCENDENTAL FUNCTION
VI	NAME FOR COMPONENTS OF INITIAL VELOCITY
V2	SQUARE OF INITIAL SPEED
X	BATTIN ITERATION VARIABLE

BATCON Analysis

BATCON is a conic propagator using the Battin universal variable formulation. A total derivation is too involved to be given here; rather the results of Battin's work will be given here.

Let the initial state of a point mass moving under the influence of a gravitational force μ be given by $\vec{r}_o$, $\vec{v}_o$. It is required to determine the state $\vec{r}$, $\vec{v}$ at a time T units later. It is useful to introduce the parameters

$$\sigma_o = \frac{\vec{r}_o \cdot \vec{v}_o}{\sqrt{\mu}} \qquad (1)$$

$$\alpha = \frac{2}{r_o} - \frac{v_o^{2}}{\mu}$$

Battin's approach is to introduce a new independent variable x(t) in place of time by the relation

$$\frac{dx}{dt} = \frac{\sqrt{\mu}}{r(t)} \qquad x(0) = 0 \qquad (2)$$

This parametrization greatly simplifies the conic propagation problem. For suppose that the value of x corresponding to t = T is given by X, i.e. x(T) = X. Then the final state is given by

$$\vec{r} = R_1(X)\,\vec{r}_o + R_2(X)\,\vec{v}_o \qquad (3)$$

$$\vec{v} = V_1(X)\,\vec{r}_o + V_2(X)\,\vec{v}_o$$

where

$$R_1(X) = 1 - \frac{1}{r_o}U_2(X) \qquad R_2(X) = \frac{1}{\sqrt{\mu}}\left[r_o U_1(X) + \sigma_o U_2(X)\right] \qquad (4)$$

$$V_1(X) = -\frac{\sqrt{\mu}}{r_o\, r_o}U_1(X) \qquad V_2(X) = 1 - \frac{1}{r_o}\quad U_2(X)$$

and where

$$
\begin{aligned}
U_0(X) &= \cos\ \alpha X \qquad \alpha > 0 \qquad & U_1(X) &= \frac{\sin\sqrt{\alpha}\ X}{\sqrt{\alpha}} \qquad \alpha > 0 \\
&= \cosh\sqrt{-\alpha}\ X \qquad \alpha < 0 & &\ \frac{\sinh\sqrt{-\alpha}\ X}{\sqrt{-\alpha}} \qquad \alpha < 0 \\
U_2(X) &= \frac{1-U_0(X)}{\alpha} & U_3(X) &= \frac{X-U_1(X)}{\alpha}
\end{aligned}
\tag{5}
$$

The problem is thus reduced to the determination of X. X is generated iteratively by the recursive formulae

$$
x_{n+1} = x_n - \frac{\sqrt{\mu}\ t_n - \sqrt{\mu}\ t}{r_n} = x_n - \Delta x \tag{6}
$$

where

$$
\begin{aligned}
\sqrt{\mu}\ \ t_n &= r_o U_1(x_n) + \sigma U_2(x_n) + U_3(x_n) \\
r_n &= r_o U_o(x_n) + \sigma U_1(x_n) + U_2(x_n)
\end{aligned}
\tag{7}
$$

To start the process the initial guess is set to

$$
x_o = \frac{\sqrt{\mu}\ T}{r_o} \left\{ 1 - \frac{\sigma_o}{2r_o^{\ 2}}\sqrt{\mu}\ T + \frac{1}{6r_o^{\ 4}} \left[3\sigma_o^{\ 2} - r_o(1-\alpha r_o) \right] \mu T^2 \right\} \tag{8}
$$

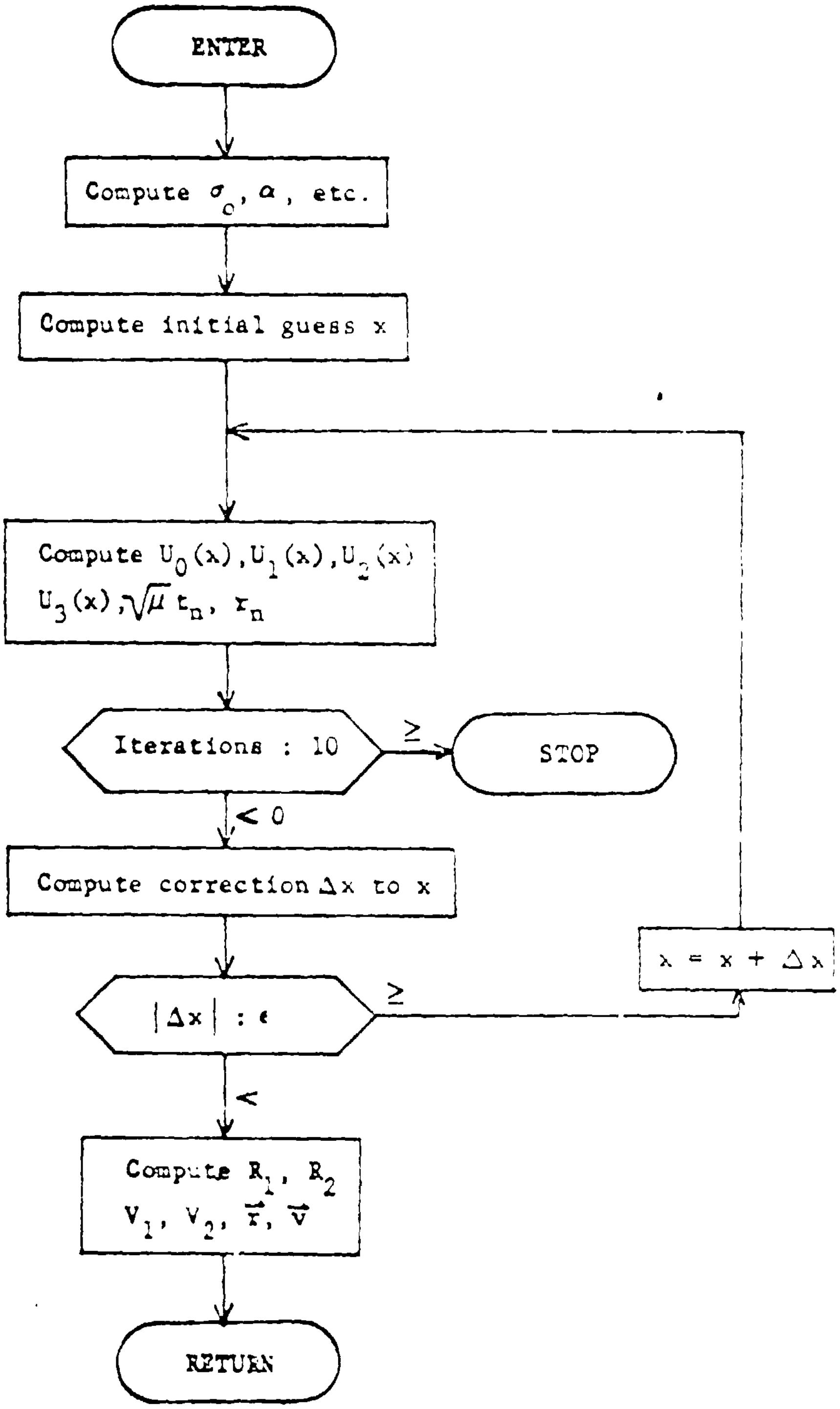
The program sets $X = x_n$ when the correction Δx is less than 10^{-8}. It terminates if the number of iterations exceeds 10.

References:

Battin, R.H., Astronautical Guidance, McGraw-Hill Book Co., New York, 1964.

Battin, R. H. and Fraser, D.C., Space Guidance and Navigation, AIAA Professional Study Series, 1970.

BATCON Flowchart



SUBROUTINE BIAIM

PURPOSE: TO PERFORM BIASED AIMPOINT GUIDANCE.

CALLING SEQUENCE: CALL BIAIM(RI,TEVN)

ARGUMENTS:	RI	I	NOMINAL SPACECRAFT STATE AT TIME OF BIASED AIMPOINT GUIDANCE EVENT
	TEVN	I	TIME OF BIASED AIMPOINT GUIDANCE EVENT

SUBROUTINES SUPPORTED: GUISIM GUIDM

SUBROUTINES REQUIRED: MATIN POLCOM PSIM GCOMP

LOCAL SYMBOLS	ADA1	VARIATION MATRIX AT TIME T(J+1)
	BB	RIGHT HALF PARTITION OF ADA1 MATRIX
	CSQ	CONSTANT DEFINING CONSTRAINT ELLIPSE
	C1	A COEFFIEIENT IN THE NECESSARY CONDITION
	C2	A COEFFIEIENT IN THE NECESSARY CONDITION
	C3	A COEFFIEIENT IN THE NECESSARY CONDITION
	C4	A COEFFIEIENT IN THE NECESSARY CONDITION
	C5	A COEFFIEIENT IN THE NECESSARY CONDITION
	C	SQUARE ROOT OF CSQ
	DELMU	AIMPOINT BIAS IN IMPACT PLANE
	DENOM	INTERMEDIATE VARIABLE
	DET	DETERMINANT OF PROJECTED TARGET CONDITION COVARIANCE MATRIX
	DVBIAS	BIAS VELOCITY CORRECTION
	DVTT	TOTAL VELOCITY CORRECTION IF BIAS IS REMOVED
	DVT	TOTAL VELOCITY CORRECTION IF BIAS IS APPLIED
	DVUPP	UPDATE VELOCITY ITERATE
	D1	PARTIAL DERIVATIVE USED IN NEWTON ITERATION TECHNIQUE

D2	PARTIAL DERIVATIVE USED IN NEWTON ITERATION TECHNIQUE
D3	PARTIAL DERIVATIVE USED IN NEWTON ITERATION TECHNIQUE
D4	PARTIAL DERIVATIVE USED IN NEWTON ITERATION TECHNIQUE
EE	MATRIX DEFINING FUNCTION TO BE MINIMIZED
IKNT	COUNTER ON NEWTON ITERATION LOOP
IS	INDEX OF NEXT GUIDANCE EVENT
ITRN	COUNTER ON OUTER ITERATION LOOP
NDIM1S	STORAGE FOR NDIM1
NDIM2S	STORAGE FOR NDIM2
PHII	INVERSE OF STATE TRANSITION MATRIX
PHI1	INTERMEDIATE ARRAY
PSIJ1	GUIDANCE MATRIX PSI AT T(J+1)
PSIJ	GUIDANCE MATRIX PSI AT EVENT TIME T(J)
QUOT	INTERMEDIATE VARIABLE
RF	DUMMY VECTOR
SAVET	STORAGE FOR TRTM1
SUM1	INTERMEDIATE VARIABLE
SUM	INTERMEDIATE VARIABLE
TWOE	CONSTANT DEFINING CONSTRAINT ELLIPSE
VCA	SPACECRAFT CLOSEST APPROACH VELOCITY RELATIVE TO TARGET PLANET
XK4	INTERMEDIATE VARIABLE
XMU	MOST RECENT IMPACT PLANE AIMPOINT
XM1	AIMPOINT ITERATE
XM2	AIMPOINT ITERATE
XM	STORAGE FOR MOST RECENT AIMPOINT ITERATE

XN1	NEGATIVE OF CONSTRAINT EQUATION EVALUATED AT MOST RECENT AIMPOINT ITERATE
XN2	NEGATIVE OF NECESSARY CONDITION EVALUATED AT MOST RECENT AIMPOINT ITERATE
YY	INTERMEDIATE VARIABLE
ZH	AIMPOINT INCREMENT FOR MOST RECENT ITERATION
ZK	AIMPOINT INCREMENT FOR MOST RECENT ITERATION

COMMON COMPUTED/USED:	A	CR	DVN	DVRB	DVUP
	EXEC	IBIAS	IIGP	PHI2	RCA
	TMPR	TRTH1	XMUS		
COMMON COMPUTED:	DELTH				
COMMON USED:	ADA	ALNGTH	ATRANS	DUMMYQ	EM3
	IDENS	IEND	IGUID	II	ISTMC
	ITR	NTP	ONE	PHI	PMASS
	POI	PROBI	RADIUS	TINJ	TM
	TNOMB	TNOMC	TWO	T3	VINF
	XLAMI	XLAM	ZERO		

BIAIM Analysis

Subroutine BIAIM performs biased aimpoint guidance computations. If planetary quarantine constraints are in effect at injection or at a midcourse correction, and if the nominal aimpoint does not satisfy these constraints, subroutine BIAIM will compute a biased aimpoint and the required bias velocity correction such that the constraints are satisfied and some performance functional is minimized.

Aimpoint biasing is performed in the impact plane and as such permits only two degrees of freedom in the selection of the biased aimpoint. The general aimpoint in the impact plane will be denoted by the 2-dimensional vector $\vec{\mu}_j$, where the j-subscript indicates that the biased aimpoint guidance event is occurring at time t_j. Three midcourse guidance policies are available in STEAP, and it will be necessary to relate $\vec{\mu}_j$ to the specific aimpoint for each of these three policies. These relationships are summarized below:

(a) Two-variable B-plane (2VBP):

$$\vec{\mu}_j = \begin{bmatrix} B \cdot T \\ B \cdot R \end{bmatrix} \tag{1}$$

(b) Three-variable B-plane (3VBP):

$$\vec{\mu}_j = \begin{bmatrix} 1 & 0 & 0 \\ 0 & 1 & 0 \end{bmatrix} \begin{bmatrix} B \cdot T \\ B \cdot R \\ t_{SI} \end{bmatrix} \tag{2}$$

(c) Fixed-time-of-arrival (FTA):

$$\vec{\mu}_j = A \, \vec{r}_{CA} \tag{3}$$

where $\vec{r}_{CA}$ is the nominal closest approach position of the spacecraft relative to the target planet. Coordinate transformation A projects the 3-dimensional vector $\vec{r}_{CA}$ (referred to ecliptic coordinates) into an equivalent FTA impact plane which is defined to be the plane containing $\vec{r}_{CA}$ and perpendicular to the spacecraft closest approach velocity $\vec{v}_{CA}$ relative to the target planet. If the ecliptic coordinates of $\vec{r}_{CA}$ and $\vec{v}_{CA}$ are denoted by r_x, r_y, r_z and v_x, v_y, v_z, respectively, then the transformation A is given by

$$A = \begin{bmatrix} \dfrac{r_x}{r_{CA}} & \dfrac{r_y}{r_{CA}} & \dfrac{r_z}{r_{CA}} \\ \dfrac{r_y v_z - r_z v_y}{r_{CA} v_{CA}} & \dfrac{r_z v_x - r_x v_z}{r_{CA} v_{CA}} & \dfrac{r_x v_y - r_y v_x}{r_{CA} v_{CA}} \end{bmatrix} \tag{4}$$

Spacecraft state variations at t_j are related to aimpoint variations (target condition variations) by the variation matrix η_j, which is always available prior to calling BLAIM. Thus, the statistical state dispersions about the nominal following the guidance correction at t_j and represented by the control covariance $P_{c_j}^+$, can be related to the dispersions about the nominal aimpoint represented by W_j^+ according to the equation

$$W_j^+ = \eta_j \, P_{c_j}^+ \, \eta_j^T \tag{5}$$

The control covariance $P_{c_j}^+$ is computed from

$$P_{c_j}^+ = P_{k_j}^- + \left[\begin{array}{c|c} 0 & 0 \\ \hline 0 & \tilde{Q}_j \end{array}\right] \tag{6}$$

where $P_{k_j}^-$ is the knowledge covariance prior to the guidance event and $\tilde{Q}_j$ is the execution error covariance.

Transformations employed in equations (1) through (3) can also be employed to project W_j^+ into the impact plane. The resulting projection is denoted by the covariance Λ_j, and is obtained from W_j^+ according to the following equations:

(a) 2VBP : $$\Lambda_j = W_j^+ \tag{7}$$

(b) 3VBP : $$\Lambda_j = \begin{bmatrix} 1 & 0 & 0 \\ 0 & 1 & 0 \end{bmatrix} W_j^+ \begin{bmatrix} 1 & 0 \\ 0 & 1 \\ 0 & 0 \end{bmatrix} \tag{8}$$

(c) PTA : $$\Lambda_j = A \, W_j^+ \, A^T \tag{9}$$

With covariance Λ_j available, it is now possible to compute the probability of impact PØI. Assuming the probability density function associated with Λ_j is gaussian and nearly constant over the target planet capture area permits us to compute PØI using the equation

$$PØI = \pi R_c^2 \, p \tag{10}$$

where R_c is the target planet capture radius and p represents the gaussian density function evaluated at the target planet center and is given by

$$p = \frac{1}{2\pi |\Lambda_j|^{\frac{1}{2}}} \exp\left[-\frac{1}{2} \vec{\mu}^{*T} \Lambda_j^{-1} \vec{\mu}^{*}\right] \tag{11}$$

The nominal impact plane aimpoint is denoted by $\vec{\mu}^{*}$. Subroutine BIAIM calls subroutine PØICØM to perform the computations involved in equations (7) through (11).

Capture radius R_c is simply the physical radius R_p of the target planet if the FTA guidance policy is employed, while for the two B-plane policies the capture radius is given by

$$R_c = R_p \sqrt{1 + \frac{2\mu_p}{V_\infty^2 R_p}} \tag{12}$$

where μ_p is the target planet gravitational constant and V_∞ is the hyperbolic excess velocity.

If the probability of impact PØI does not exceed the permissible impact probability P_I, and if the nominal aimpoint has not been previously biased, we simply return to subroutine GUIDM (or GUISIM). If the nominal aimpoint has been previously biased, a velocity correction ΔV_{RB_j} required to remove that bias is computed prior to returning. But if PØI exceeds P_I, an aimpoint bias $\delta\vec{\mu}_j$ and the associated bias velocity correction $\Delta\vec{V}_{B_j}$ must be computed. Before describing the details of the biasing technique it is necessary to define the relationship between $\Delta\vec{V}_j$ and $\delta\vec{\mu}_j$ for linear midcourse guidance policies.

Linear impulsive guidance policies have form

$$\Delta\vec{V}_j = \Gamma_j \, \delta\hat{X}_j \tag{13}$$

where Γ_j is the guidance matrix and $\delta\vec{X}_j$ is the spacecraft state deviation from the targeted nominal trajectory. (These guidance policies are discussed in more detail in the subroutine GUIS analysis section.) Such guidance policies can be readily generalized to account for changes in the target conditions from their nominal values. This generalized version of equation (13) has form

$$\Delta\vec{V}_j = \Gamma_j \; \delta\vec{X}_j + \Psi_j \; \delta\vec{\mu}_j \tag{14}$$

where Ψ_j can also be referred to as a guidance matrix. For the purposes of the BIAIM analysis, we shall assume that $\delta\vec{\mu}_j$ in equation (14) is always an aimpoint change in the impact plane. Thus, Ψ_j will be a 3x2 guidance matrix. The derivation of the Ψ_j matrix is quite similar to the derivation of the Γ_j matrix and will not be presented here. If we partition the previously discussed variation matrix η_j as follows:

$$\eta_j = \left[\eta_1 \;\middle|\; \eta_2 \right] \tag{15}$$

then the Ψ_j matrices for the three midcourse guidance policies are given by the following equations:

(a) 2VBP :
$$\Psi_j = \eta_2^{\;T} \left(\eta_2 \, \eta_2^{\;T} \right)^{-1} \tag{16}$$

(b) 3VBP :
$$\Psi_j = \eta_2^{\;-1} \begin{bmatrix} 1 & 0 \\ 0 & 1 \\ 0 & 0 \end{bmatrix} \tag{17}$$

(c) FTA :
$$\Psi_j = \eta_2^{\;-1} A^T \tag{18}$$

If an aimpoint bias were to be removed at time t_j, the required velocity correction would be given by

$$\Delta\vec{V}_{RB_j} = - \Psi_j \; \delta\vec{\mu}_j \tag{19}$$

If an aimpoint bias were to be imparted at time t_j, the bias velocity correction would be given by

$$\Delta\vec{V}_{B_j} = \Psi_j \; \delta\vec{\mu}_j \tag{20}$$

If an aimpoint bias $\delta\hat{\mu}_j^{(1)}$ had been previously imparted, and if a new aimpoint bias $\delta\hat{\mu}_j^{(2)}$ is to be imparted, then the total bias velocity correction would be given by

$$\Delta\hat{V}_{B_j} = \Psi_j \left[\delta\hat{\mu}_j^{(2)} - \delta\hat{\mu}_j^{(1)} \right] \tag{21}$$

The general statement of the biased aimpoint guidance problem is as follows: Find an aimpoint $\hat{\mu}_j$ in the impact plane which satisfies the impact probability constraint

$$P\emptyset I \leq P_I \tag{22}$$

and minimizes a performance functional having form

$$J = (\hat{\mu}_j - \hat{\mu}^*)^T \tilde{A} (\hat{\mu}_j - \hat{\mu}^*) \tag{23}$$

where $\hat{\mu}^*$ is the nominal aimpoint and $\tilde{A}$ is a constant symmetric matrix that will be defined subsequently.

The solution of this problem is detailed in the section on biased aimpoint guidance in the analytical manual. Only the results will be presented here. The assumption of constant probability density over the target planet capture area permits us to rewrite constraint equation (22) as

$$\lambda_1 \mu_1^2 + 2\lambda_3 \mu_1 \mu_2 + \lambda_2 \mu_2^2 = C^2 \tag{24}$$

where

$$C^2 = 2 \ln \left[\frac{R_c^2}{2 \, |\Lambda|^{\frac{1}{2}} P_I} \right] \tag{25}$$

and $\hat{\mu} = \begin{bmatrix} \mu_1 \\ \mu_2 \end{bmatrix}$ and $\Lambda^{-1} = \begin{bmatrix} \lambda_1 & \lambda_3 \\ \lambda_3 & \lambda_2 \end{bmatrix}$.

The inequality has been replaced by an equality since the solution can be shown to lie on the constraint boundary, which, from inspection of equation (24) is an ellipse centered at the target planet.

If t_j is the time of the final midcourse correction, matrix $\tilde{A}$ will be chosen as a 2x2 identity matrix. The minimization of J is then equivalent to minimization of the miss distance $|\hat{\mu}_j - \hat{\mu}^*|$. If t_j is not the final midcourse correction time, $\tilde{A}$ will be defined as follows:

$$\tilde{A} = \Psi_{j+1}^{T} \Psi_{j+1} \tag{26}$$

Here Ψ_{j+1} denotes the aimpoint guidance matrix for the next midcourse correction occurring at time t_{j+1}. In this case the minimization of J is equivalent to the minimization of $\left| \Delta \vec{V}_{RB_{j+1}} \right|$, i.e., the velocity required to remove bias $\delta\vec{\mu}_j$ at time t_{j+1} will by minimized. The computation of Ψ_{j+1} is based on the variation matrix η_{j+1}, just as Ψ_j was based on η_j. However, η_{j+1} can be computed more efficiently by using the relationship

$$\eta_{j+1} = \eta_j \Phi_{j+1,\, j}^{-1} \tag{27}$$

where $\Phi_{j+1,j}$ is the state transition matrix over $\left[t_j, t_{j+1} \right]$.

If we define

$$\tilde{A} = \begin{bmatrix} a_1 & a_3 \\ a_3 & a_2 \end{bmatrix}$$

then the necessary condition for a minimum is given by

$$(a_1 \lambda_3 - a_3 \lambda_1) \mu_1^2 + (a_3 \lambda_2 - a_2 \lambda_3) \mu_2^2 + (a_1 \lambda_2 - a_2 \lambda_1) \mu_1 \mu_2$$

$$+ (-a_1 \lambda_3 \mu_1^* - a_3 \lambda_3 \mu_2^* + a_3 \lambda_1 \mu_1^* + a_2 \lambda_1 \mu_2^*) \mu_1 +$$

$$(-a_1 \lambda_2 \mu_1^* - a_3 \lambda_2 \mu_2^* + a_3 \lambda_3 \mu_1^* + a_2 \lambda_3 \mu_2^*) \mu_2 = 0 \tag{28}$$

Thus, our problem is reduced to finding μ_1 and μ_2 which satisfy equations (24) and (28). Since the analytical solution of these equations proved intractable, a standard Newton iteration technique is employed in BIAIM which quickly converges to solutions for μ_1 and μ_2. The iteration process is started with an initial guess defined as the intersection of the extended $\vec{\mu}^*$ vector and the constraint boundary defined by equation (24). This initial guess is given by

$$\mu_1^{\circ} = \left(\frac{\mu_1^*}{\mu_2^*} \right) \mu_2^{\circ}$$

$$\mu_2^{\circ} = \operatorname{sgn}(\mu_2^*) \frac{c}{\sqrt{\lambda_1 \left(\frac{\mu_1^*}{\mu_2^*} \right)^2 + 2\lambda_3 \left(\frac{\mu_1^*}{\mu_2^*} \right) + \lambda_2}} \tag{29}$$

where c is defined by equation (25).

In addition to the previously described iteration process, subroutine BIAIM also employs an outer iteration loop which accounts for the dependence of $\tilde{Q}_j$ (equation (6)) on $\delta\hat{\mu}_j$. The execution error covariance $\tilde{Q}_j$ is a function of the total velocity corre ion at t_j, but the total velocity correction, in particular $\Delta\hat{V}_{B_j}$, depends on $\delta\hat{\mu}_j$. This coupling is resolved by recomputing $\tilde{Q}_j$ at the end of the previously described biasing technique and repeating the biasing cycle until the error function

$$\left| P\text{Ø}I - P_I \right| \leq P_I \times 10^{-3}$$

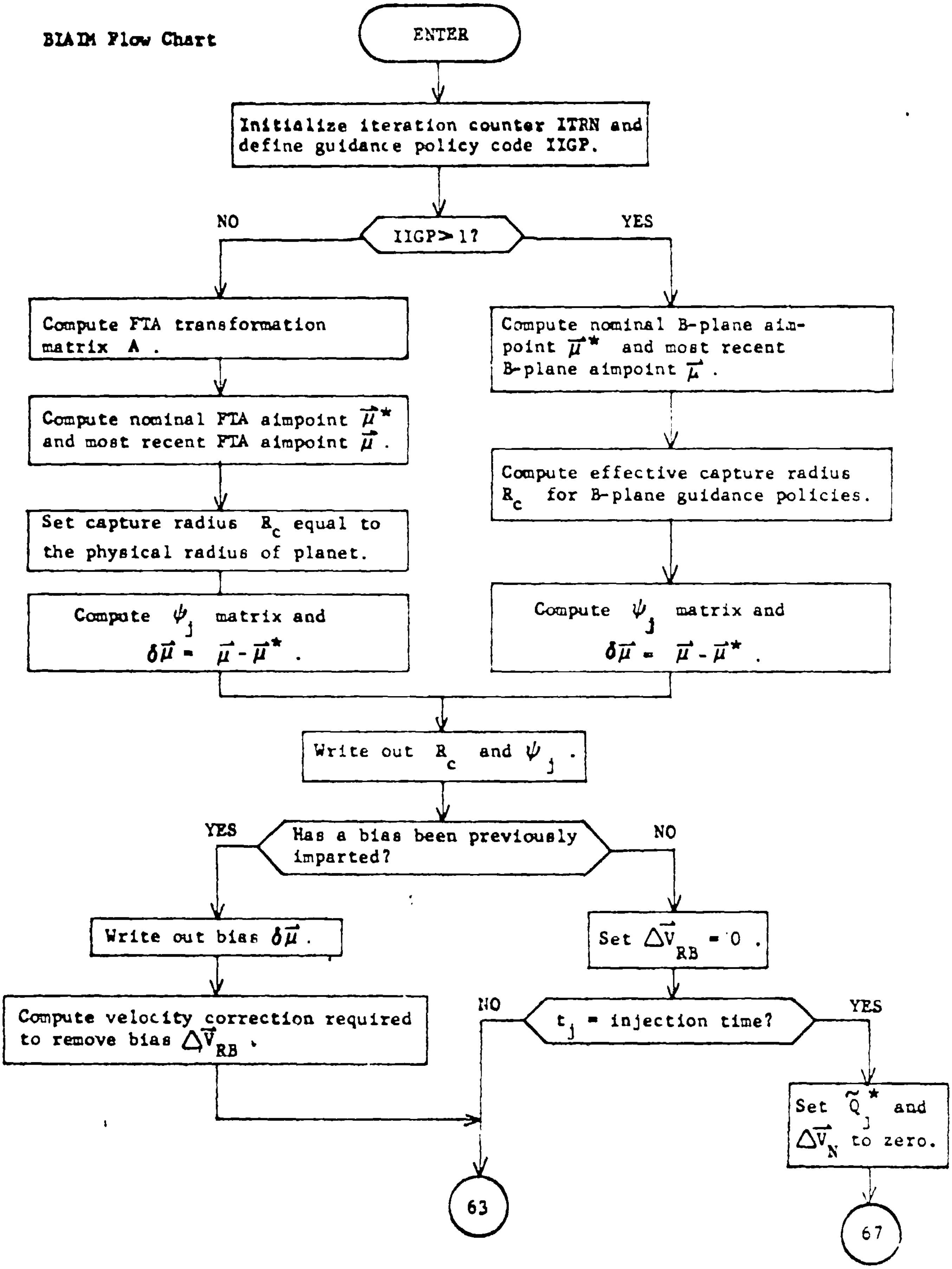
is satisfied. This outer iteration process is not performed, however, if t_j = injection time since at injection equation (6) is replaced by the equation

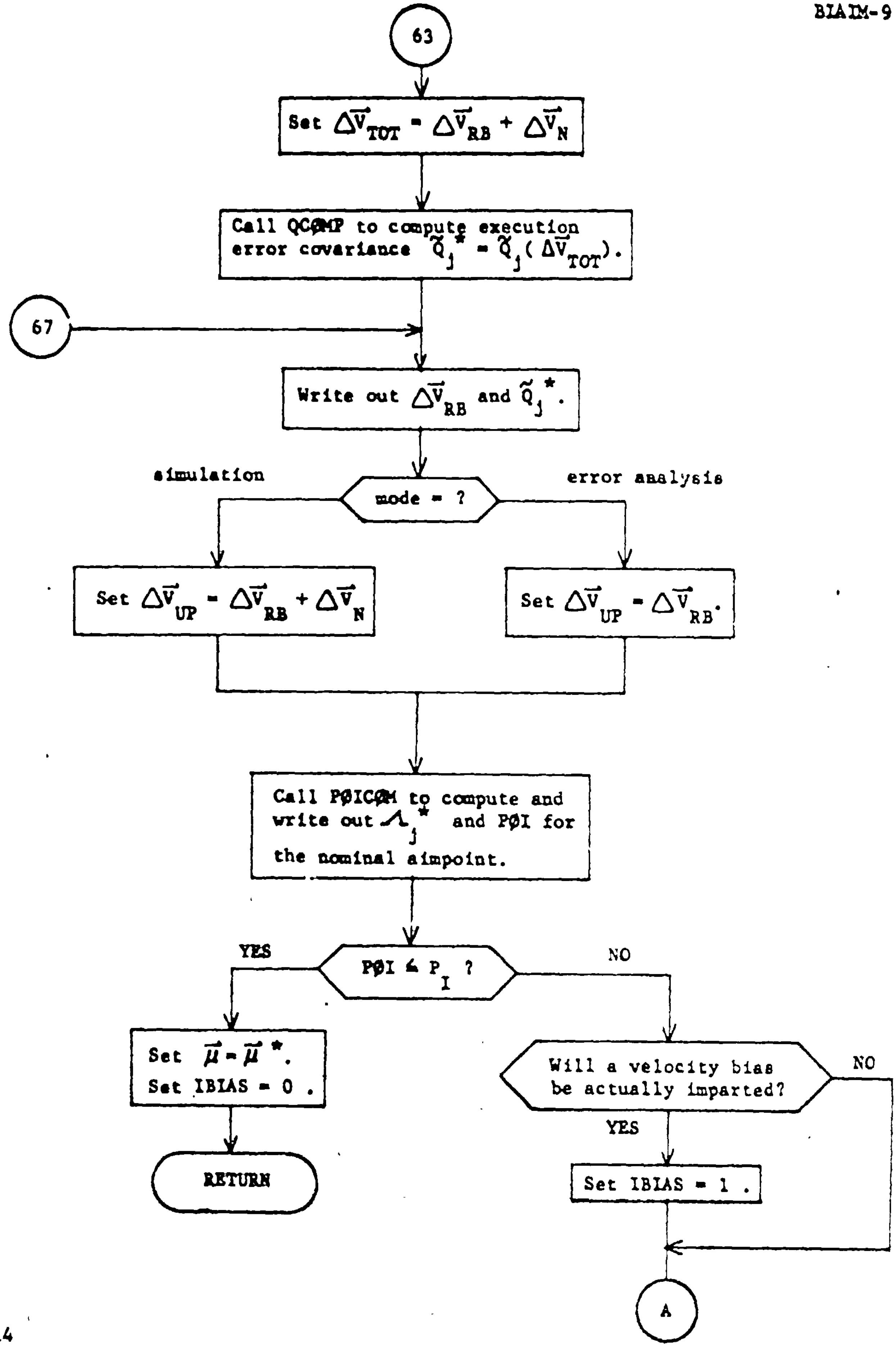
$$P_{c_j} = P_{k_j}$$

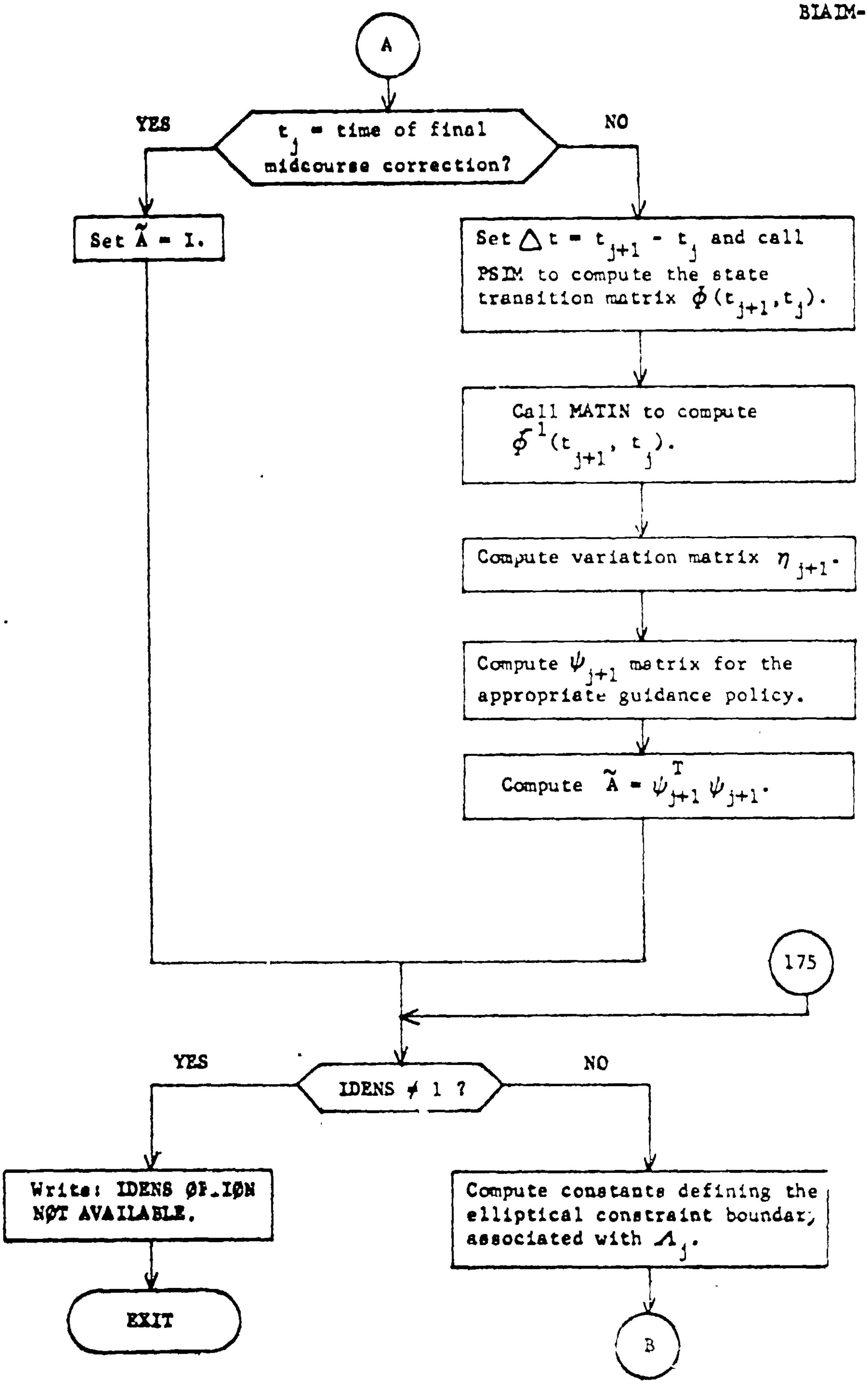
and $\tilde{Q}_j$ is always zero.

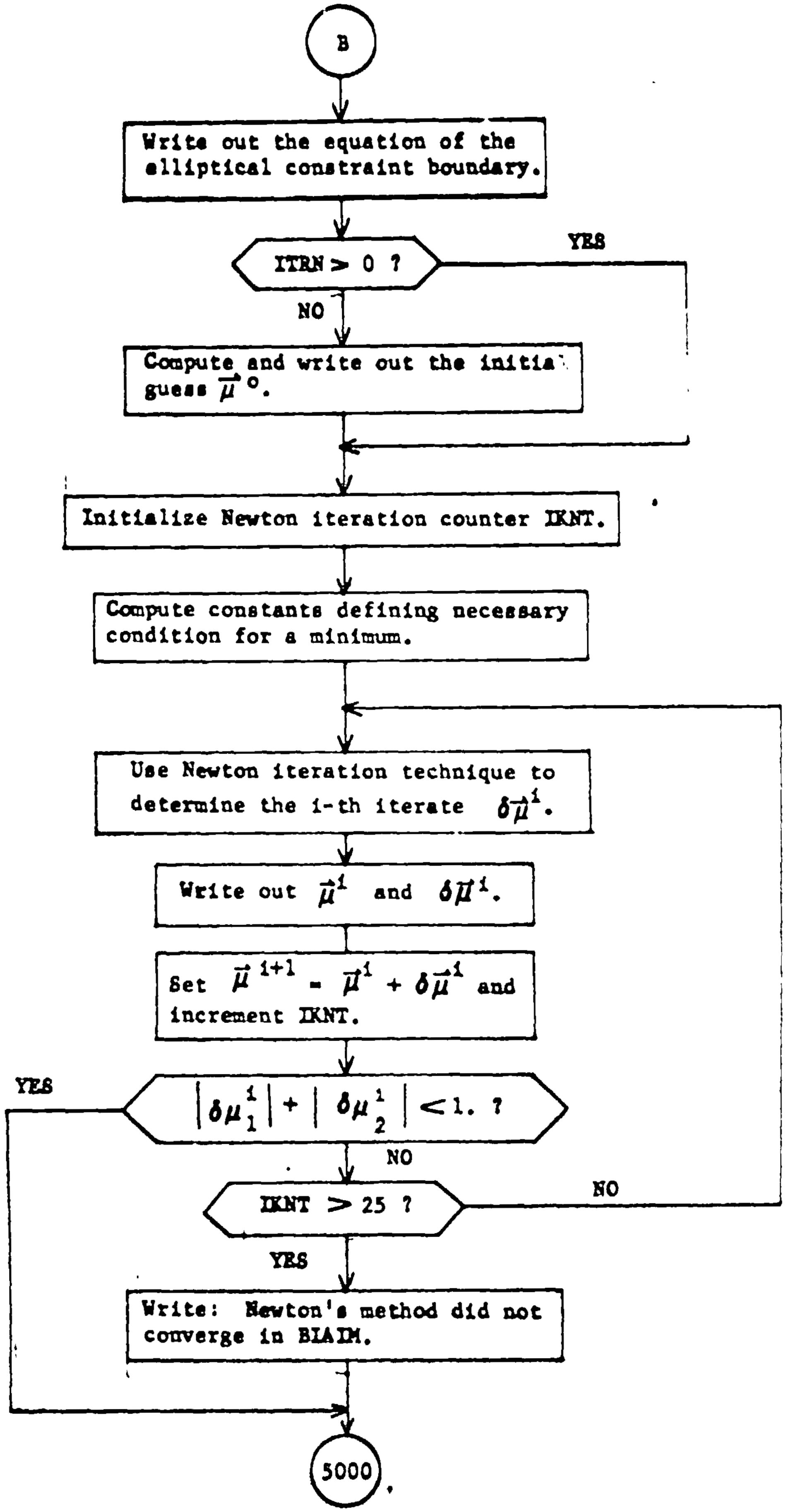
Reference: Mitchell, R. T., and Wong, S. K.: Preliminary Flight Path Analysis Orbit Determination and Maneuver Strategy Mariner Mars 1969. Project Document 138, Jet Propulsion Laboratory, 1968.

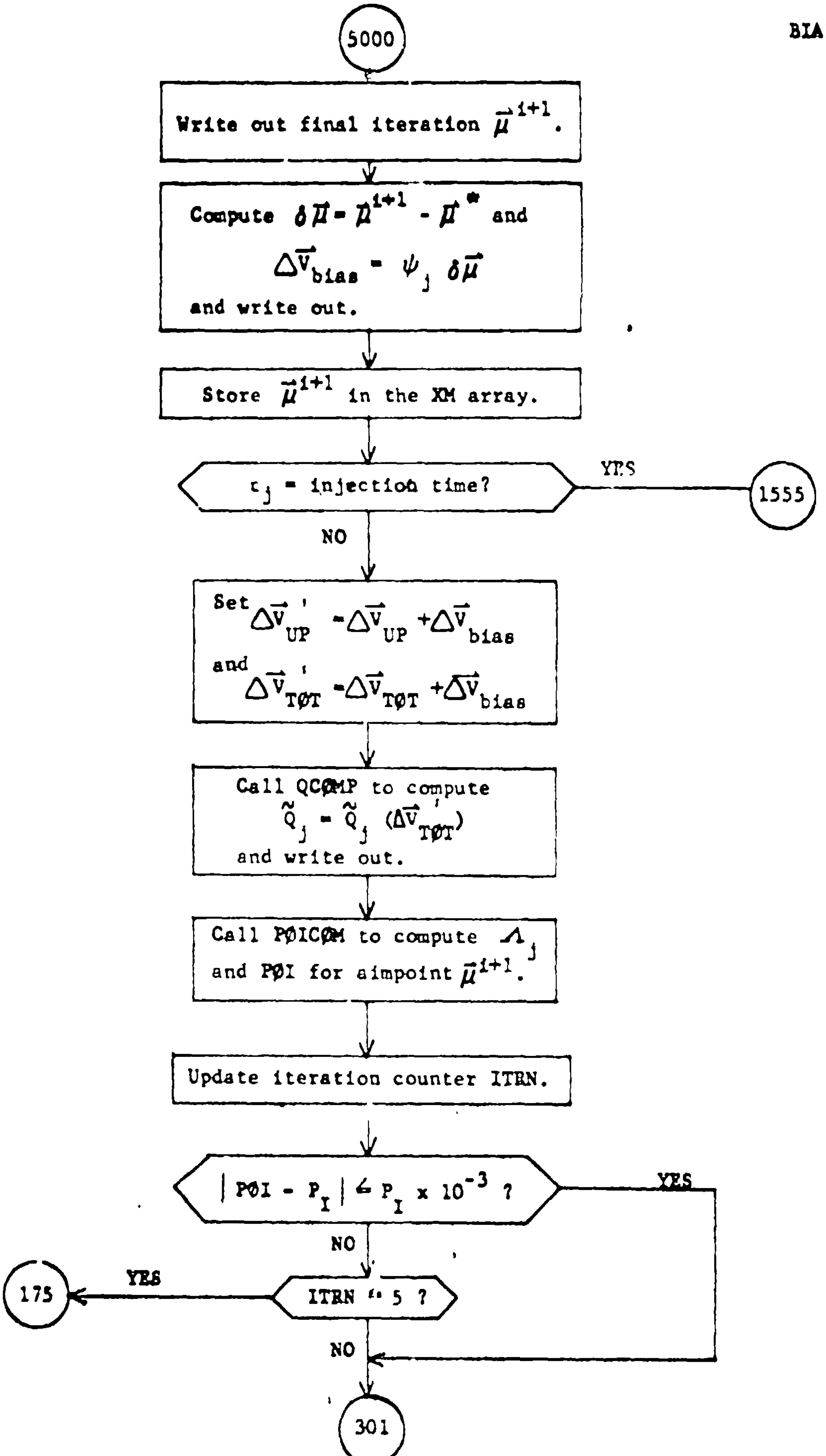
BIAIM Flow Chart

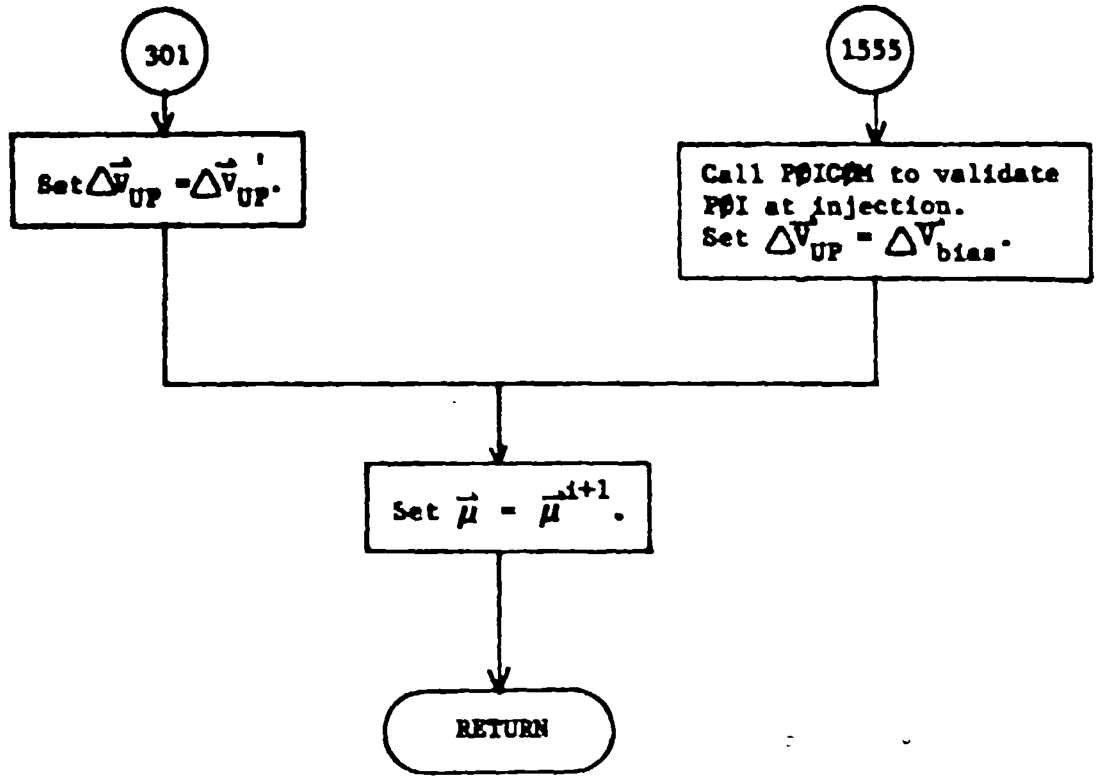



63
Set $\Delta\vec{V}_{TOT} = \Delta\vec{V}_{RB} + \Delta\vec{V}_N$
Call QCØMP to compute execution error covariance $\tilde{Q}_j^* = \tilde{Q}_j(\Delta\vec{V}_{TOT})$.
67
Write out $\Delta\vec{V}_{RB}$ and $\tilde{Q}_j^*$.
simulation
mode = ?
error analysis
Set $\Delta\vec{V}_{UP} = \Delta\vec{V}_{RB} + \Delta\vec{V}_N$
Set $\Delta\vec{V}_{UP} = \Delta\vec{V}_{RB}$.
Call PØICØM to compute and write out Λ_j^* and PØI for the nominal aimpoint.
YES
PØI $\leq P_I$?
NO
Set $\vec{\mu} = \vec{\mu}^*$.
Set IBIAS = 0 .
RETURN
Will a velocity bias be actually imparted?
NO
YES
Set IBIAS = 1 .
A


A
YES
t_j = time of final midcourse correction?
NO
Set $\tilde{A}$ = I.
Set $\Delta t = t_{j+1} - t_j$ and call PSIM to compute the state transition matrix $\phi(t_{j+1}, t_j)$.
Call MATIN to compute $\phi^{-1}(t_{j+1}, t_j)$.
Compute variation matrix η_{j+1}.
Compute ψ_{j+1} matrix for the appropriate guidance policy.
Compute $\tilde{A} = \psi_{j+1}^T \psi_{j+1}$.
175
YES
IDENS ≠ 1 ?
NO
Write: IDENS ØP.IØN NØT AVAILABLE.
EXIT
Compute constants defining the elliptical constraint boundary associated with Λ_j.
B


B
Write out the equation of the elliptical constraint boundary.
ITRN > 0 ?
YES
NO
Compute and write out the initial guess $\vec{\mu}^{\,o}$.
Initialize Newton iteration counter IKNT.
Compute constants defining necessary condition for a minimum.
Use Newton iteration technique to determine the i-th iterate $\delta\vec{\mu}^{\,i}$.
Write out $\vec{\mu}^{\,i}$ and $\delta\vec{\mu}^{\,i}$.
Set $\vec{\mu}^{\,i+1} = \vec{\mu}^{\,i} + \delta\vec{\mu}^{\,i}$ and increment IKNT.
YES
$|\delta\mu_1^i| + |\delta\mu_2^i| < 1.$?
NO
IKNT > 25 ?
NO
YES
Write: Newton's method did not converge in BIAIM.
5000


5000
Write out final iteration $\vec{\mu}^{i+1}$.
Compute $\delta\vec{\mu} = \vec{\mu}^{i+1} - \vec{\mu}^{*}$ and
$\Delta\vec{V}_{bias} = \psi_j \, \delta\vec{\mu}$
and write out.
Store $\vec{\mu}^{i+1}$ in the XM array.
t_j = injection time?
YES
1555
NO
Set $\Delta\vec{V}_{UP}{}' = \Delta\vec{V}_{UP} + \Delta\vec{V}_{bias}$
and $\Delta\vec{V}_{TØT}{}' = \Delta\vec{V}_{TØT} + \Delta\vec{V}_{bias}$
Call QCØMP to compute
$\tilde{Q}_j = \tilde{Q}_j \, (\Delta\vec{V}_{TØT}{}')$
and write out.
Call PØICØM to compute Λ_j
and PØI for aimpoint $\vec{\mu}^{i+1}$.
Update iteration counter ITRN.
$|PØI - P_I| \le P_I \times 10^{-3}$?
YES
NO
ITRN $\ge$ 5 ?
YES
175
NO
301


301
1555
Set $\Delta\vec{V}_{UP} = \Delta\vec{V}_{UP}'$.
Call PØICØM to validate
PØI at injection.
Set $\Delta\vec{V}_{UP} = \Delta\vec{V}_{bias}$.
Set $\vec{\mu} = \vec{\mu}^{i+1}$.
RETURN

SUBROUTINE BIAS

PURPOSE: COMPUTE THE ACTUAL MEASUREMENT BIAS IN THE SIMULATION PROGRAM

RETURN THE ACTUAL MEASUREMENT BIAS TO BE USED IN THE SIMULATION MODE.

CALLING SEQUENCE: CALL BIAS(MCODE,BVAL)

ARGUMENT: BVAL O THE ACTUAL BIAS TO BE USED IN THE MEASUREMENT

MCODE I MEASUREMENT TYPE CODE

SUBROUTINES SUPPORTED: SIMULL

COMMON USED: BIA

BIAS Analysis

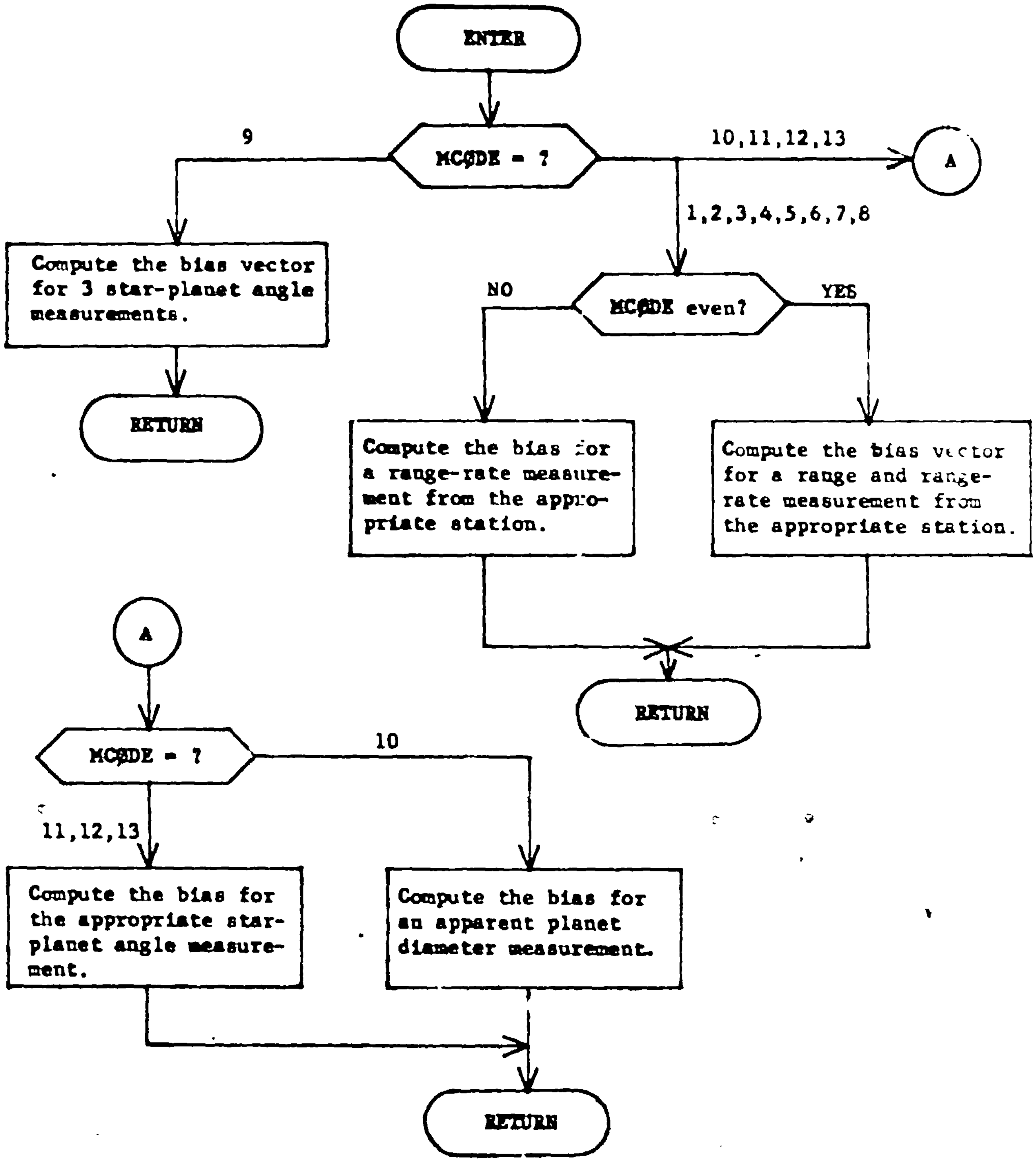
The actual measurement Y_k^a at time t_k is given by

$$Y_k^a = \underline{Y}_k + b_k + \nu_k$$

where $\underline{Y}_k$ is the ideal measurement, which would be made in the absence of instrumentation errors, b_k is the actual measurement bias and ν_k represents the actual measurement noise.

The function of subroutine BIAS is to compute the measurement bias b_k for the appropriate measurement type. The constant biases for all measurement devices are stored in the vector BIA. Subroutine BIAS selects the appropriate elements from this vector to construct the actual measurement bias.

BIAS Flow Chart



BLOCK DATA

PURPOSE: TO LOAD CONSTANTS INTO COMMON LOCATIONS USED IN VARIOUS OTHER PARTS OF THE PROGRAM.

CALLING SEQUENCE: NONE

ARGUMENT: NONE

SUBROUTINES SUPPORTED HALF THE SUBROUTINES USE THE CONSTANTS STORED BY THIS SUBROUTINE

SUBROUTINES REQUIRED NONE

COMMON LOADED					
	CN1	CN	ELMNT	EMN	EVNM
	RADIUS	RAD	RMASS	SMJR	SPHERE
	F	MNNAME	PI	PLANET	PMASS
	ST				

BLKDAT Analysis

Subroutine BLKDAT is responsible for setting up constants used in computing ephemeris data for the gravitating bodies.

The arrays set up by BLKDAT and their definitions are as follows:

Array	Definition
CN(80)	Constants defining mean elements for inner planets
ST(50)	Constants defining mean elements for outer planets
SMJR(18)	Constants defining semi-major axes for planets and moon
EMN(15)	Constants defining lunar elements
PMASS(11)	Gravitational constants of sun, planets, and moon
RMASS(11)	Mass of bodies relative to sun
RADIUS(11)	Surface radii of sun, planets, and moon
SPHERE(11)	Sphere of influence radii of sun, planets, and moon
MONTH(12)	Names of months for output purposes
PLANET(11)	Names of planets for output purposes

The definitions of the CN, ST, SMJR, and EMN arrays are provided in Tables 2 through 5 on the following page. The actual constants stored in those arrays are the ephemeris data listed on the next pages following.

The constants stored in the other arrays are given below.

Body	PMASS (AU^3/day^2)	RMASS*	RADIUS (AU)	SPHERE (AU)
Sun	2.959122083(-4)	1.0	4.66582(-3)	NA
Mercury	4.850(-11)	1.639(-7)	1.617(-5)	7.46(-4)
Venus	7.243(-10)	2.448(-6)	4.044(-5)	4.12(-3)
Earth	8.88757(-10)	3.003(-6)	4.263(-5)	6.18(-3)
Mars	9.5497905(-11)	3.236(-7)	2.279(-5)	3.78(-3)
Jupiter	2.8252(-7)	9.547(-4)	4.7727(-4)	.3216
Saturn	8.454(-8)	2.857(-4)	4.0374(-4)	.3246
Uranus	1.290(-8)	4.359(-5)	1.5761(-4)	.346
Neptune	1.5(-8)	5.069(-5)	1.4906(-4)	.5805
Pluto	7.4(-10)	2.501(-6)	4.679(-5)	.2366
Moon	1.0921748(-11)	3.696(-8)	1.161(-5)	3.71394(-4)

* Truncated from program values

Array Definitions

Constant	i	Ω	$\tilde{\omega}$	e	M	a	ω	E
Mercury	1	2	3	4	5	6	7	8
Venus	9	10	11	12	13	14	15	16
Earth	17	18	19	20	21	22	23	24
Mars	25	26	27	28	29	30	31	32
Jupiter	33	34	35	36	37	38	39	40
Saturn	41	42	43	44	45	46	47	48
Uranus	49	50	51	52	53	54	55	56
Neptune	57	58	59	60	61	62	63	64
Pluto	65	66	67	68	69	70	71	72
Moon	73	74	75	76	77	78	79	80

Table 1. ELMNT Array -- Conic Elements

a_0	a_1
1	2
3	4
5	6
7	8
9	10
11	12
13	14
15	16
17	18

Table 2. SMJR Array

Constant	i_0	i_1	i_2	i_3	Ω_0	Ω_1	Ω_2	Ω_3	$\tilde{\omega}_0$	$\tilde{\omega}_1$	$\tilde{\omega}_2$	$\tilde{\omega}_3$	e_0	e_1	e_2	e_3	M_0	M_1	M_2	M_3
Mercury	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16	17	18	19	20
Venus	21	22	23	24	25	26	27	28	29	30	31	32	33	34	35	36	37	38	39	40
Earth	41	42	43	44	45	46	47	48	49	50	51	52	53	54	55	56	57	58	59	60
Mars	61	62	63	64	65	66	67	68	69	70	71	72	73	74	75	76	77	78	79	80

Table 3. CM Array -- Inner Planet Constants

Constant	i_0	i_1	Ω_0	Ω_1	$\tilde{\omega}_0$	$\tilde{\omega}_1$	e_0	e_1	M_0	M_1
Jupiter	1	2	3	4	5	6	7	8	9	10
Saturn	11	12	13	14	15	16	17	18	19	20
Uranus	21	22	23	24	25	26	27	28	29	30
Neptune	31	32	33	34	35	36	37	38	39	40
Pluto	41	42	43	44	45	46	47	48	49	50

Table 4. ST Array -- Outer Planet Constants

Constant	Ω_0	Ω_1	Ω_2	Ω_3	$\tilde{\omega}_0$	$\tilde{\omega}_1$	$\tilde{\omega}_2$	$\tilde{\omega}_3$	L_0	L_1	L_2	L_3	i	e	a
Moon	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15

Table 5. EMN Array -- Lunar Constants

Planetary and Lunar Ephemerides

Mean Elements of Mercury

$$i = 0.1222233228 + 3.24776685 \times 10^{-5}\ T - 3.199770295 \times 10^{-7}\ T^2$$

$$\Omega = 0.8228518595 + 2.068578774 \times 10^{-2}\ T + 3.034933644 \times 10^{-6}\ T^2$$

$$\tilde{\omega} = 1.3246996178 + 2.714840259 \times 10^{-2}\ T + 5.143873156 \times 10^{-6}\ T^2$$

$$e = 0.20561421 + 0.00002046\ T - 0.000000030\ T^2$$

$$M = 1.785111955 + 7.142471000 \times 10^{-2}\ d + 8.72664626 \times 10^{-9}\ D^2$$

$$a = 0.3870986 \text{ A.U.} = 57{,}909{,}370 \text{ km}$$

Mean Elements of Venus

$$i = 0.0592300268 + 1.7555510339 \times 10^{-5}\ T - 1.696847884 \times 10^{-8}\ T^2$$

$$\Omega = 1.3226043500 + 1.570534527 \times 10^{-2}\ T + 7.155849933 \times 10^{-6}\ T^2$$

$$\tilde{\omega} = 2.2717874591 + 2.457486613 \times 10^{-2}\ T + 1.704120089 \times 10^{-5}\ T^2$$

$$e = 0.00682069 - 0.00004774\ T + 0.000000091\ T^2$$

$$M = 3.710626172 + 2.796244623 \times 10^{-2}\ d + 1.682497399 \times 10^{-6}\ D^2$$

$$a = 0.7233316 \text{ A.U.} = 108{,}209{,}322 \text{ km}$$

Mean Elements of Earth

$$i = 0$$

$$\Omega = 0$$

$$\tilde{\omega} = 1.7666368138 + 3.000526417 \times 10^{-2}\ T + 7.902463002 \times 10^{-6}\ T^2 + 5.817764173 \times 10^{-8}\ T^3$$

$$e = 0.01675104 - 0.00004180\ T - 0.000000126\ T^2$$

$$M = 6.256583781 + 1.720196977 \times 10^{-2}\ d - 1.954768762 \times 10^{-7}\ D^2 - 1.22173047 \times 10^{-9}\ D^3$$

$$a = 1.0000003 \text{ A.U.} = 149{,}598{,}530 \text{ km}$$

Mean Elements of Mars

$$i = 0.0322944089 - 1.178097245 \times 10^{-5}\ T + 2.201054112 \times 10^{-7}\ T^2$$

$$\Omega = 0.8514840375 + 1.345634309 \times 10^{-2}\,T - 2.424068406 \times 10^{-8}\,T^2$$
$$- 9.308422677 \times 10^{-8}\,T^3$$

$$\tilde{\omega} = 5.8332085089 + 3.212729365 \times 10^{-2}\,T + 2.266503959 \times 10^{-6}\,T^2$$
$$-2.084698829 \times 10^{-8}\,T^3$$

$$e = 0.09331290 + 0.000092064\,T - 0.000000077\,T^2$$

$$M = 5.576840523 + 9.145887726 \times 10^{-3}\,d + 2.365444735 \times 10^{-7}\,D^2$$
$$+ 4.363323130 \times 10^{-10}\,D^3$$

$$a = 1.5236915 \text{ A.U.} = 227{,}941{,}963 \text{ km}$$

Mean Elements of Jupiter

$$i = 0.0228410270 - 9.696273622 \times 10^{-5}\,T$$

$$\Omega = 1.7355180770 + 1.764479392 \times 10^{-2}\,T$$

$$\tilde{\omega} = 0.2218561704 + 2.812302353 \times 10^{-2}\,T$$

$$e = 0.0483376 + 0.00016302\,T$$

$$M = 3.93135411 + 1.450191928 \times 10^{-3}\,d$$

$$a = 5.202803 \text{ A.U.} = 778{,}331{,}525 \text{ km}$$

Mean Element of Saturn

$$i = 0.0435037861 - 7.757018898 \times 10^{-8}\,T$$

$$\Omega = 1.9684445802 + 1.523977870 \times 10^{-2}\,T$$

$$\tilde{\omega} = 1.5897996653 + 3.419861162 \times 10^{-2}\,T$$

$$e = 0.0558990 - 0.00034705\,T$$

$$M = 3.0426210430 + 5.837120844 \times 10^{-4}\,d$$

$$a = 9.538843 \text{ A.U.} = 1{,}426{,}996{,}160 \text{ km}$$

Mean Elements of Uranus

$$i = 0.0134865470 + 0.696273622 \times 10^{-6}\,T$$

$$\Omega = 1.2826407705 + 8.912087493 \times 10^{-3}\,T$$

$$\tilde{\omega} = 2.9502426085 + 2.834608631 \times 10^{-2}\,T$$

$e = 0.0470463 + 0.00027204\ T$

$M = 1.2843599198 + 2.046548840 \times 10^{-4}\ d$

$a = (19.182281 - 0.00037008\ T)$ A.U. $= (2{,}869{,}640{,}310 - 85271\ T)$ km

Mean Elements of Neptune

$i = 0.0310537707 - 1.599885148 \times 10^{-4}\ T$

$\Omega = 2.2810642235 + 1.923032859 \times 10^{-2}\ T$

$\tilde{\omega} = 0.7638202701 + 1.532704516 \times 10^{-2}\ T$

$e = 0.00852849 + 0.00007701\ T$

$M = 0.7204851506 + 1.033089473 \times 10^{-4}\ d$

$a = (30.057053 + 0.001210166\ T)$ A.U. $= (4{,}496{,}490{,}000 + 181039\ T)$ km

Mean Elements of Pluto

$i = 0.2996706970859694$

$\Omega = 1.1914337550102258$

$\tilde{\omega} = 3.909919302791948$

$e = 0.2488033053623924$

$M = 3.993890007 + 0.6962635708298997 \times 10^{-4}$

$a = 39.37364135300176$ A.U. $= 5{,}890{,}213{,}786{,}145{,}730$ km

Mean Elements of Moon

$i = 5.1453964^{\circ}$

$\Omega = 259.183275^{\circ} - 0.0529539222d + 0.002078\ T^2 + 0.000002\ T^3$

$\tilde{\omega} = 334.329556^{\circ} + 0.1114040803d - 0.010325\ T^2 - 0.000012\ T^3$

$L = 270.434164^{\circ} + 13.1763965268d - 0.001133\ T^2 + 0.0000019\ T^3$

$a = .00256954448$ A.U.

$e = 0.054900489$

Note 1: The above elements are referred to the mean equinox and ecliptic of date except for Pluto.

Note 2: The elements for pluto are oscillating values for epoch 1960 September 23.0 E.T. = J.D. 2437200.5

Note 3: The time interval from the epoch is denoted by T when measured in Julian centuries of 36,525 ephemeris days, by D = 3.6525 T when measured in units of 10,000 ephemeris days, and by d = 10,000D = 36,525 T when measured in ephemeris days. Times are measured with respect to the epoch 1900 January 0.5 E.T. = J.D. 2415020.0.

Note 4: Angular relations are expressed in radians for planets and degrees for moon.

References: (1) Space Research Conic Program, Phase III, J.P.L., May 1969 (Planetary constants)

(2) The American Ephemeris and Nautical Almanac - 1965, U.S. Government Printing Office, Washington, p. 493 (Lunar constants)

SUBROUTINE CAREL

PURPOSE: TRANSFORM CARTESIAN COORDINATES TO CONIC ELEMENTS

CALLING SEQUENCE: OALL CAREL(GM,R,V,TFP,A,E,W,XI,XN,TA,PP,QQ,WW)

ARGUMENT:	GM	I	GRAVITATIONAL CONSTANT OF THE CENTRAL BODY
	R(3)	I	POSITION VECTOR RELATIVE TO CENTRAL BODY
	V(3)	I	VELOCITY VECTOR RELATIVE TO CENTRAL BODY
	TFP	O	TIME OF FLIGHT FROM PERIAPSIS ON THE CONIC
	A	O	SEMI-MAJOR AXIS OF THE CONIC
	E	O	ECCENTRICITY OF THE CONIC
	W	O	ARGUMENT OF PERIAPSIS OF THE CONIC
	XI	O	INCLINATION OF THE CONIC TO THE REFERENCE FRAME
	XN	O	LONGITUDE OF THE ASC(NDING NODE OF THE CONIC
	TA	O	INSTANTANEOUS TRUE ANOMALY OF THE CONIC
	PP(3)	O	UNIT VECTOR TOWARD PERIAPSIS ON CONIC
	QQ(3)	O	UNIT VECTOR NORMAL TO PP IN ORBITAL PLANE
	WW(3)	O	UNIT VECTOR NORMAL TO ORBITAL PLANE

SUBROUTINES SUPPORTED: TAROPT LUNCON MULTAR EXCUTE COPINS
NONINS CPROP VMP GUISIM NONLIN
PULSEX GUIDM

SUBROUTINES REQUIRED: NONE

LOCAL SYMBOLS:	AUXF	ECCENTRIC ANOMALY (HYPERBOLIC CASE)
	AVA	MEAN ANOMALY (ELLIPTIC CASE)
	COSEA	COSINE OF THE ECCENTRIC ANOMALY (ELLIPTIC CASE)
	CTA	COSINE OF THE TRUE ANOMALY
	C	MAGNITUDE OF THE ANGULAR MOMENTUM
	DIV	INTERMEDIATE VARIABLE IN CALCULATION OF ECCENTRIC ANOMALY

EA	ECCENTRIC ANOMALY (ELLIPTIC CASE)
P	SEMI-LATUS RECTUM OF THE CONIC
RAD	DEGREES TO RADIANS CONVERSION CONSTANT
RD	TIME DERIVATIVE OF RADIUS
RM	MAGNITUDE OF CARTESIAN POSITION VECTOR
SINEA	SINE OF THE ECCENTRIC ANOMALY (ELLIPTIC CASE)
SINHF	HYPERBOLIC SINE OF AUXF
STA	SINE OF THE TRUE ANOMALY
TANG	INTERMEDIATE VARIABLE USED TO CALCULATE SINHF
VM	MAGNITUDE OF THE CARTESIAN VELOCITY VECTOR
Z	INTERMEDIATE VECTOR USED TO CALCULATE PP, QQ VECTORS

CAREL Analysis

CAREL converts the cartesian state (position and velocity) of a massless point referenced to a gravitational body to the equivalent conic elements about that body.

Let the cartesian state be denoted $\vec{r}, \vec{v}$ and let the gravitational constant of the central body be μ.

The angular momentum constant c is

$$c = \left| \vec{r} \times \vec{v} \right| \tag{1}$$

The unit normal $\hat{W}$ to the orbital plane is

$$\hat{W} = \frac{\vec{r} \times \vec{v}}{c} \tag{2}$$

The semilatus rectum p is

$$p = \frac{c^2}{\mu} \tag{3}$$

The semi-major axis a is

$$a = \frac{r}{2 - \frac{rv^2}{\mu}} \tag{4}$$

Thus $a > 0$ for elliptical motion, $a < 0$ for hyperbolic motion. The eccentricity e is

$$e = \sqrt{1 - \frac{p}{a}} \tag{5}$$

Thus $e < 1$ for elliptical motion, $e > 1$ for hyperbolic motion. The inclination of the orbit i is computed from

$$\cos i = \hat{W}_z \tag{6}$$

The longitude of the ascending node Ω is defined by

$$\tan \Omega = \frac{\hat{W}_x}{-\hat{W}_y} \tag{7}$$

The true anomaly f at the given state is computed from

$$\cos f = \frac{p - r}{e\,r} \qquad \sin f = \frac{c\dot{r}}{\mu e} \tag{8}$$

Now define an auxiliary vector $\hat{z}$ by

$$\hat{z} = \frac{r}{c}\,\vec{v} - \frac{\dot{r}}{c}\,\vec{r} \tag{9}$$

Then $\hat{P}$, the unit vector to periapsis, and $\hat{Q}$, the in-plane normal to $\hat{P}$, are defined by

$$\hat{P} = \hat{r}\cos f - \hat{z}\sin f \tag{10}$$

$$\hat{Q} = \hat{r}\sin f + \hat{z}\cos f \tag{11}$$

where $\hat{r} = \frac{\vec{r}}{r}$. The argument of periapsis ω is then computed from

$$\tan\omega = \frac{\hat{P}_z}{\hat{Q}_z} \tag{12}$$

The conic time from periapsis t_p is computed from different formulae depending upon the sign of the semi-major axis. For $a > 0$ (elliptical motion)

$$t_p = \sqrt{\frac{a^3}{\mu}}\,(E - e\sin E)$$

$$\cos E = \frac{e + \cos f}{1 + e\cos f} \qquad \sin E = \frac{\sqrt{1 - e^2}\,\sin f}{1 + e\cos f} \tag{13}$$

For $a < 0$ (hyperbolic motion) the time from periapsis is

$$t_p = \sqrt{\frac{a^3}{\mu}}\,(e\sinh H - H)$$

$$\tanh\frac{H}{2} = \sqrt{\frac{e - 1}{e + 1}}\,\tan\frac{f}{2} \tag{14}$$

Reference: Battin, R. H., Astronautical Guidance, McGraw-Hill Book Co., New York, 1964.

SUBROUTINE CASCAD

PURPOSE: TO COMPUTE THE STATE TRANSITION MATRIX DEFINING STATE PERTURBATIONS OVER AN ARBITRARY TIME INTERVAL BY CASCADING DANBY MATRIZANTS OVER SEGMENTS OF THE INTERVAL USING EITHER PATCHED CONIC OR VIRTUAL MASS TWO BODY FORMULAE.

CALLING SEQUENCE: CALL CASCAD(RI,STMAT)

ARGUMENT: RI I POSITION AND VELOCITY OF VEHICLE AT BEGINNING OF TIME INTERVAL

STMAT O STATE TRANSITION MATRIX OVER DESIRED INTERVAL

SUBROUTINES SUPPORTED: PSIM

SUBROUTINES REQUIRED: CONC2 VMP

LOCAL SYMBOLS:

DELTAT	TIME INTERVAL USED IN A SINGLE ITERATE
DELT	TIME INTERVAL OF CURRENT PROPAGATION
D1	INITIAL TIME OF ITERATE
IFLAG	FLAG TO DETERMINE WHETHER ITERATION IS COMPLETED
IOS	FLAG INDICATING HELIOCENTRIC OR PLANETOCENTRIC PHASE
ISP3	FLAG USED AS TRAJECTORY INTEGRATION SPHERE OF INFLUENCE STOPPING CODE
PHI	CUMULATIVE STATE TRANSITION MATRIX OVER INTERVAL (T0, TK)
PSI	CUMULATIVE STATE TRANSITION MATRIX OVER INTERVAL (T0, TK+1)
PTP	STATE OF TARGET RELATIVE TO INERTIAL COORDINATE AT TIME TK
RAV	STATE OF SPACECRAFT RELATIVE TO DOMINANT BODY FOR MATRIZANT
RHO	STATE TRANSITION MATRIX OVER INTERVAL (TK,TK+1)
RS	INERTIAL SPACECRAFT STATE AT TK
RSF	INERTIAL SPACECRAFT STATE AT TK+1

R1	SPACECRAFT STATE RELATIVE TO VIRTUAL MASS AT TK
R2	SPACECRAFT STATE RELATIVE TO VIRTUAL MASS AT TK+1
SUM	INTERMEDIATE VARIABLE
TIME	CUMULATIVE TRAJECTORY TIME FROM INITIAL TIME TO TK+1
XMU	VIRTUAL MASS MAGNITUDE AT TK
YMU	VIRTUAL MASS MAGNITUDE AT TK+1

COMMON COMPUTED: ICL

COMMON USED:

ACC	ALNGTH	DATEJ	DELTM	DTPLAN
DTSUM	ISTM1	NTP	PMASS	RTP
RVS	TM	TRTM1	VMU	V

CASCAD Analysis

CASCAD approximates the state transition matrix $\Phi_{f,o}$ defining state perturbations over an arbitrary interval $\left[t_o, t_f\right]$ by recursively computing state transition matrices over intervals $\left[t_o, t_1\right]$, $\left[t_o, t_2\right]$, ... , $\left[t_o, t_f\right]$.

The recursive formula for the k+1 iteration based on the k-th iteration is given by

$$\Phi_{k+1,o} = \Psi_{k+1,k}\ \Phi_{k,o} \tag{1}$$

where $\Psi_{k+1,k}$ is the state transition matrix for the k+1-st interval $\left[t_k, t_{k+1}\right]$.

The time interval $\Delta t_{k+1} = t_{k+1} - t_k$ is determined by the position vector $\vec{r}_k$ of the spacecraft relative to the target planet along the nominal n-body trajectory at the time t_k. Then if R_{SOI} denotes the radius of the sphere of influence of the target planet the time interval is defined by

$\Delta t_{k+1} = \Delta t_{planet}$		if $r_k \leq R_{SOI}$
$= \Delta t_{sun}$		if $r_k > R_{SOI}$ and the n-body nominal trajectory propagated over Δt_{sun} does not intersect the SOI.
$= \Delta t_{SOI}$		if $r_k > R_{SOI}$ and the n-body nominal trajectory intersects the SOI after the time interval Δt_{SOI} where Δt_{SOI} Δt_{sun}.

where Δt_{planet} and Δt_{sun} are input parameters. For the last interval a partial step may be required so that $\Delta t_n = t_f - t_{n-1}$.

The $\Psi_{k+1,k}$ matrix may be computed by either of two models. In the patch conic model the position and velocity vectors $\vec{R}_k$, $\vec{V}_k$ of the spacecraft relative to the dominant body (the sun if $\Delta t_{k+1} = \Delta t_{sun}$ or Δt_{SOI}, the target planet if $\Delta t_{k+1} = \Delta t_{planet}$) at the time t_k is used to define a

conic with respect to the dominant body and the Danby matrizant over the given interval defines $\Psi_{k+1,k}$ (CØNC2) .

In the virtual mass model the position and velocity vectors $\vec{R}_k$, $\vec{V}_k$ are computed relative to the virtual mass and the gravitational constant used is that of the virtual mass magnitude at the time t_k. The Danby matrizant corresponding to this conic then is used to compute $\Psi_{k+1,k}$ (CØNC2).

The recursive process continues until the state transition matrix over the entire interval $[t_o, t_f]$ is determined.

Reference: Danby, J.M.A., "The Matrizant of Keplerian Motion," AIAA Journal, vol 2, no 1, January, 1964.

SUBROUTINE CENTER

PURPOSE: TO CONVERT THE POSITION AND VELOCITY VECTORS OF THE GRAVITATING BODIES FROM REFERENCE BODY ECLIPTIC TO BARYCENTRIC ECLIPTIC AND STORE THEM IN THE F ARRAY.

CALLING SEQUENCE: CALL CENTER

SUBROUTINES SUPPORTED: EPHEM

SUBROUTINES REQUIRED: NONE

LOCAL SYMBOLS

BARYC	POSITION AND VELOCITY OF CENTER OF MASS RELATIVE TO EARTH. (AU, AU/DAY)	
F	ARRAY OF PLANET EPHEMERIS DATA IN AU, AU/DAY UNITS. DATA INDICATED BY THE FOLLOWING INDICES	
	4*I-2,J	VELOCITY OF I-TH PLANET RELATIVE TO THE SUN (INPUT) AND RELATIVE TO THE BARYCENTER (OUTPUT)
	4*I-3,J	POSITION OF I-TH PLANET RELATIVE TO THE SUN (INPUT) AND RELATIVE TO THE BARYCENTER (OUTPUT)
	4*IM-2,J	VELOCITY OF MOON RELATIVE TO EARTH (INPUT) AND RELATIVE TO BARYCENTER OUTPUT
	4*IM-3,J	POSITION OF MOON RELATIVE TO EARTH (INPUT) AND RELATIVE TO BARYCENTER(OUTPUT)
GEOP	POSITION AND VELOCITY OF BODIES RELATIVE TO THE EARTH. (AU, AU/DAY)	
IND	INDEX USED TO EXTRACT EARTH EPHEMERIS DATA RELATIVE TO SUN FROM F-ARRAY	
IX	INDEX OF IJ-TH GRAVITATIONAL BODY	
SUM	SUM OF GRAVITATIONAL CONSTANTS (AU**3/DAY**2)	
SUN	POSITION AND VELOCITY OF SUN RELATIVE TO EARTH (AU, AU/DAY)	

COMMON COMPUTED/USED: F INITAL V

COMMON USED: NBODYI NO PMASS ZERO

CENTER Analysis

Let the state vector of position and velocity of the gravitating bodies (excluding the moon) in heliocentric ecliptic coordinates be denoted ρ_i, ω_i at some reference time. Let the index of the earth be i_E. Then the coordinates of all bodies (excluding the moon) relative to the earth is

$$\begin{aligned} \vec{r}_i &= \vec{\rho}_i - \vec{\rho}_{i_E} \qquad i=1,n \;,\quad i \neq i_M \\ \vec{v}_i &= \vec{\omega}_i - \vec{\omega}_{i_E} \qquad i=1,n \;,\quad i \neq i_M \end{aligned} \tag{1}$$

Let the position and velocity of the moon relative to the earth be denoted $\vec{r}_{i_M}, \vec{v}_{i_M}$.

Define the radius vector to the center of mass (in earth ecliptic coordinates) by

$$R_{CM} = \frac{1}{M} \sum_{i=1}^{n} \mu_i \vec{r}_i \qquad M = \sum_{i=1}^{n} \mu_i \tag{2}$$

Its velocity relative to the earth may then be found by differentiation.

$$\vec{V}_{CM} = \frac{1}{M} \sum_{i=1}^{n} \mu_i \vec{v}_i \tag{3}$$

The coordinates of all gravitating bodies relative to the center of mass may then be computed

$$\begin{aligned} \vec{R}_i &= \vec{r}_i - \vec{R}_{CM} \\ \vec{V}_i &= \vec{v}_i - \vec{V}_{CM} \end{aligned} \tag{4}$$

SUBROUTINE CONC2

PURPOSE: COMPUTE STATE TRANSITION MATRIX USING ANALYTICAL PATCHED CONIC OR ANALYTICAL VIRTUAL MASS TECHNIQUES

CALLING SEQUENCE: CALL CONC2(R,V,DELT,GMX,PSIEC)

ARGUMENT:

	DELT	I	TIME INCREMENT OVER WHICH THE STATE TRANSITION MATRIX IS BEING COMPUTED
	GMX	I	GRAVITATIONAL CONSTANT OF GOVERNING BODY
	PSIEC	O	STATE TRANSITION MATRIX
	R	I	POSITION OF THE VEHICLE RELATIVE TO THE GOVERNING BODY
	V	I	VELOCITY OF THE VEHICLE RELATIVE TO THE GOVERNING BODY

SUBROUTINES SUPPORTED: PSIM CASCAD PCTM

LOCAL SYMBOLS:

A	SEMI-MAJOR AXIS
A1	INTERMEDIATE VARIABLE
A2	INTERMEDIATE VARIABLE
A3	INTERMEDIATE VARIABLE
AM2	INTERMEDIATE VARIABLE
C1	MAGNITUDE OF RXV
CSE	COSINE OF ECCENTRIC ANOMALY
CTA	COSINE OF TRUE ANOMALY
CTA2	COSINE OF TRUE ANOMALY ON ELLIPSE
DDXO	INTERMEDIATE VARIABLE
DDYO	INTERMEDIATE VARIABLE
DXO	INTERMEDIATE VARIABLE
DYO	INTERMEDIATE VARIABLE
E	ECCENTRICITY
EA	ECCENTRIC ANOMALY

FMI1	INTERMEDIATE VECTOR
FMI	INTERMEDIATE VECTOR
F	INTERMEDIATE VARIABLE
N	INTERMEDIATE VARIABLE
OPEC	INTERMEDIATE VECTOR
ORB	INTERMEDIATE VARIABLE
P	SEMI-LATUS RECTUM
PI	MATHEMATICAL CONSTANT
PSIOP	INTERMEDIATE STATE TRANSITION MATRIX
PV	INTERMEDIATE VECTOR
Q	INTERMEDIATE VECTOR
RD	R DOT V DIVIDED BY MAGNITUDE OF R
RM	MAGNITUDE OF R
RRD	R DOT V
RTMD	INTERMEDIATE VARIABLE
R2	INTERMEDIATE VARIABLE
R3	INTERMEDIATE VARIABLE
SNE	SINE OF ECCENTRIC ANOMALY
SNF	SINE OF F
STA	SINE OF TRUE ANOMALY
STA2	SINE OF TRUE ANOMALY ON ELLIPSE
TIM1	INTERMEDIATE TIME
TIM2	INTERMEDIATE TIME
VM	MAGNITUDE OF V
WV	RXV
XO	INTERMEDIATE VARIABLE
YO	INTERMEDIATE VARIABLE

	Z	INTERMEDIATE VECTOR				
COMMON USED:		EM8	HALF	ONE	THREE	TWOPI
		TWO	ZERO			

CONC2 Analysis

CONC2 is responsible for the computation of a state transition matrix about a conic trajectory using the Danby matrizant analytic formula.

Danby has shown (see Reference 2) that the state transition matrix (or matrizant) has a particularly simple form if written in the orbital plane coordinate system. The state transition matrix ϕ defined by

$$\delta x_f = \phi(t_f, t_o)\, \delta X_o \qquad (1)$$

where δx_f, δx_o refer to perturbations about a conic trajectory at time t_f, t_o respectively may be written in the orbital plane system

$$\bar{\phi}(t_f, t_o) = M(t_f)\, M^{-1}(t_o) \qquad (2)$$

where $M(t)$, $M^{-1}(t)$ may be computed from the following formulae

$$M = \begin{bmatrix} \dot{X} & Y\dot{X}-h & 0 & 2X-3\tau\dot{X} & Y\dot{Y} & 0 \\ \dot{Y} & -X\dot{X} & 0 & 2Y-3\tau\dot{Y} & -Y\dot{X}-2h & 0 \\ 0 & 0 & Y & 0 & 0 & -X \\ \ddot{X} & \dot{Y}\dot{X}+Y\ddot{X} & 0 & -\dot{X}-3\tau\ddot{X} & \dot{Y}^2+Y\ddot{Y} & 0 \\ \ddot{Y} & -\dot{X}^2-X\ddot{X} & 0 & -\dot{Y}-3\tau\ddot{Y} & -\dot{X}\dot{Y}-Y\ddot{X} & 0 \\ 0 & 0 & \dot{Y} & 0 & 0 & -\dot{X} \end{bmatrix} \qquad (3)$$

$$M^{-1} = AJM^TJ^T \qquad (4)$$

where $X, Y, \dot{X}, \dot{Y}, \ddot{X}, \ddot{Y}$ are evaluated at the time t
h is the angular momentum constant
τ is the time interval from t to some epoch (periapsis)

and $A = \text{diag}\,(a/\mu,\ a/\mu h,\ 1/h,\ a/\mu,\ a/\mu h,\ 1/h)$ (5)

$$J = \begin{bmatrix} 0 & -I \\ I & 0 \end{bmatrix} \qquad (6)$$

Thus to use the Danby formulation one must determine the transformation from the reference frame to the orbital plane coordinates, compute the values of the quantities $X, Y, \dot{X}, \dot{Y}, \ddot{X}, \ddot{Y}$ and h and τ at the times t_o, t_f and then use the above equations.

Let the initial state of the conic be denoted $\bar{r}$, $\bar{v}$, the gravitational force μ, and the time interval Δt. Then the unit vectors $\hat{P}$ in the direction of periapsis, $\hat{W}$ in the direction of the angular momentum vector, and $\hat{Q} = \hat{W} \times \hat{P}$ defining the orbital plane coordinate system may be computed by the following conic equations

$$h = |\vec{r} \times \vec{v}| \tag{7}$$

$$\hat{W} = \frac{\vec{r} \times \vec{v}}{h} \tag{8}$$

$$\dot{r} = \frac{\vec{r} \cdot \vec{v}}{v} \tag{9}$$

$$p = \frac{h^2}{\mu} \tag{10}$$

$$a = \frac{r}{2 - rv^2/\mu} \tag{11}$$

$$e = \sqrt{1 - p/a} \tag{12}$$

$$\cos f = \frac{p - r}{er} \qquad \sin f = \frac{\dot{r}\,h}{\mu e} \tag{13}$$

$$\vec{z} = \frac{r}{h}\vec{v} - \frac{\dot{r}}{h}\vec{r} \tag{14}$$

$$\hat{P} = \cos f \,\frac{\vec{r}}{r} - \sin f \,\vec{z} \tag{15}$$

$$\hat{Q} = \sin f \,\frac{\vec{r}}{r} + \cos f \,\vec{z} \tag{16}$$

$$\dot{f} = \frac{c}{r^2} \tag{17}$$

The transformation matrix from the original $\vec{r}$, $\vec{v}$ system to the orbital plane system may then be written

$$T = \left[\, \hat{P} \;\vdots\; \hat{Q} \;\vdots\; \hat{W} \,\right] \tag{18}$$

Let the true anomaly at the pertinent time (t_o or t_f) be denoted f. Then the quantities required in (3) are written

$$\begin{aligned} X &= r\cos f & Y &= r\sin f \\ \dot{X} &= \dot{r}\cos f - r\dot{f}\sin f & \dot{Y} &= \dot{r}\sin f + r\dot{f}\cos f \\ \ddot{X} &= -\frac{\mu X}{r^3} & \ddot{Y} &= -\frac{\mu Y}{r^3} \end{aligned} \tag{19}$$

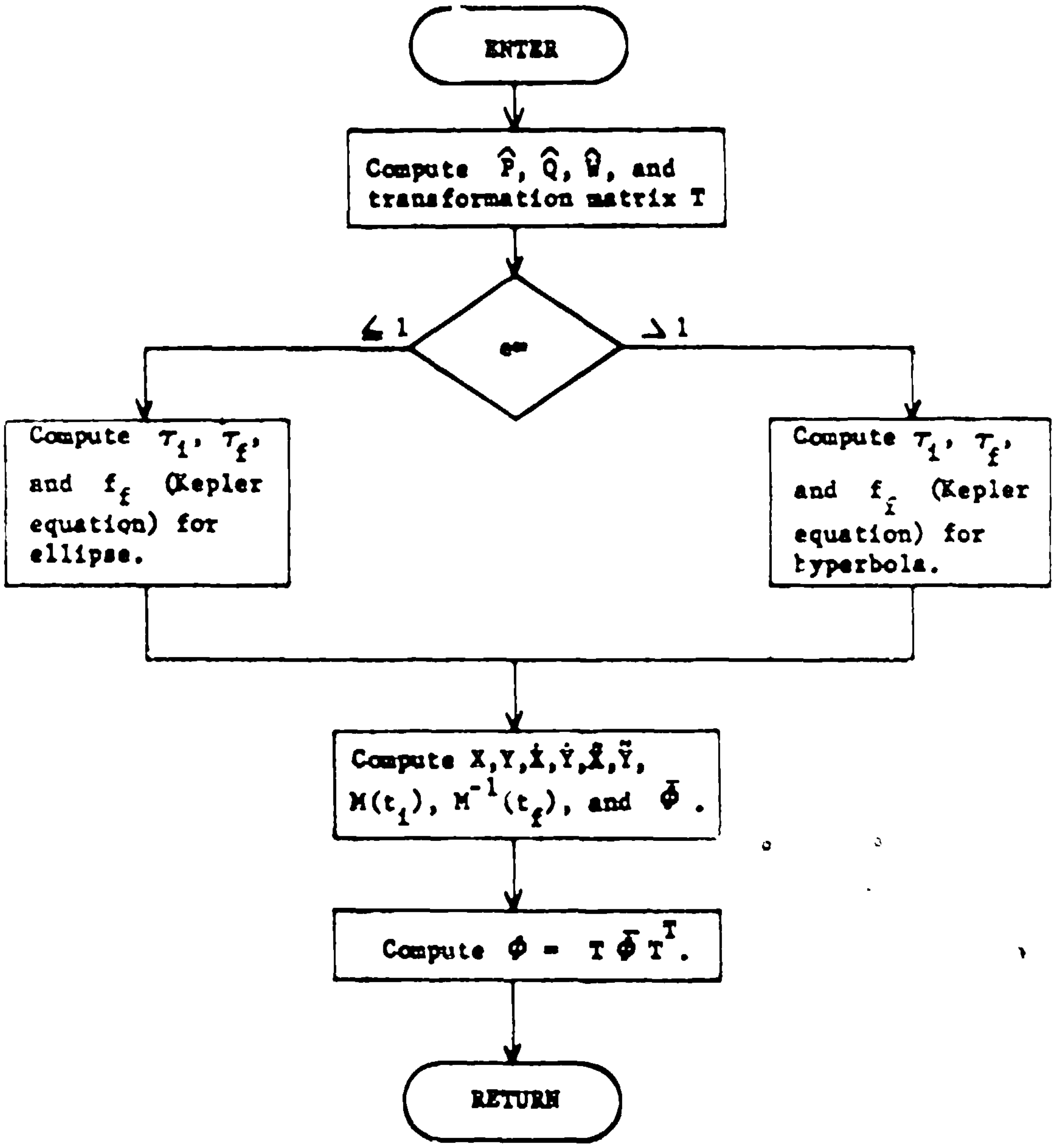
Having computed the state transition matrix $\bar{\Phi}$ corresponding to the orbital plane system by equations (2), (3), (4), it is an easy task to convert it to the normal reference system

$$\Phi = T\bar{\Phi}T^T \tag{20}$$

References: Battin, R. H., Astronautical Guidance, McGraw-Hill Book Co., New York, 1964.

Danby, J.M.A., Matrizant of Keplerian Motion, AIAA J., vol. 3, no. 4, April, 1965.

CONC2 Flow Chart



SUBROUTINE CONVRT

PURPOSE: TO COMPUTE THE GEOCENTRIC EQUATORIAL COORDINATES OF THE VEHICLE.

CALLING SEQUENCE: CALL CONVRT(R,PHI,THETA,VEL,GAMMA,SIGMA,X,Y,Z,
VX,VY,VZ)

ARGUMENT:

GAMMA	I	PATH ANGLE
R	I	GEOCENTRIC RADIUS
PHI	I	DECLINATION
THETA	I	RIGHT ASCENSION
VEL	I	VELOCITY
SIGMA	I	AZIMUTH
X	O	X COMPONENT OF POSITION IN GEOCENTRIC EQUITORIAL COORDINATES
Y	O	Y COMPONENT OF POSITION IN GEOCENTRIC EQUITORIAL COORDINATES
Z	O	Z COMPONENT OF POSITION IN GEOCENTRIC EQUITORIAL COORDINATES
VX	O	X COMPONENT OF VELOCITY IN GEOCENTRIC EQUITORIAL COORDINATES
VY	O	Y COMPONENT OF VELOCITY IN GEOCENTRIC EQUITORIAL COORDINATES
VZ	O	Z COMPONENT OF VELOCITY IN GEOCENTRIC EQUITORIAL COORDINATES

SUBROUTINES SUPPORTED: DATA DATAS

LOCAL SYMBOLS:

B1	INTERMEDIATE VARIABLE
B2	INTERMEDIATE VARIABLE
B3	INTERMEDIATE VARIABLE
CG	COSINE OF PATH ANGLE
CP	COSINE OF DECLINATION
CT	COSINE OF RIGHT ASCENSION
SG	SINE OF PATH ANGLE

SP	SINE OF DECLINATION
ST	SINE OF RIGHT ASCENSION

CONVRT Analysis

Geocentric equatorial position and velocity components are related to geocentric radius, declination, right ascension, velocity magnitude, flight path angle, and azimuth through the following equations:

$$x = r \cos \phi \cos \theta$$

$$y = r \cos \phi \sin \theta$$

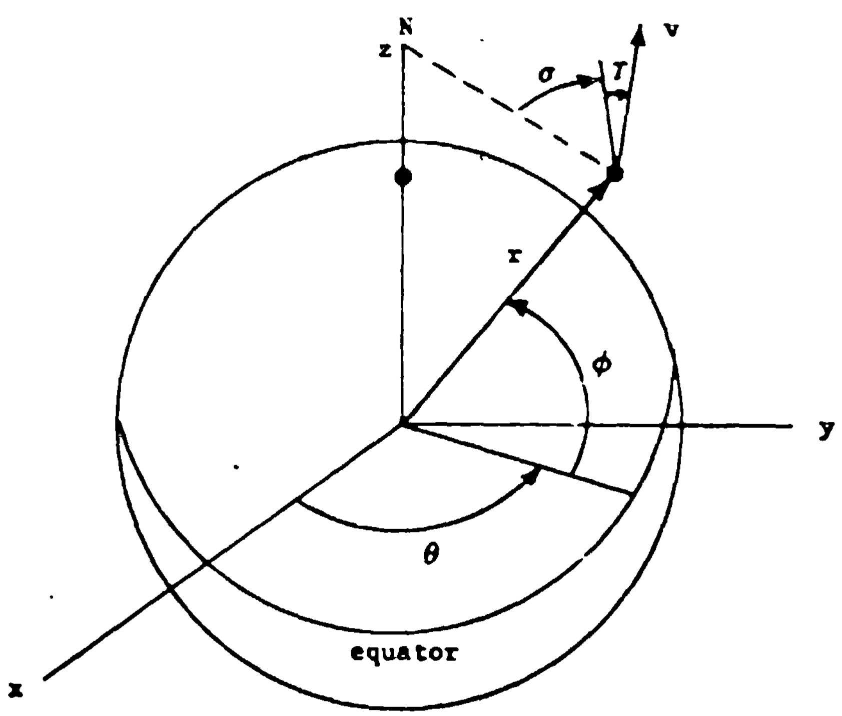
$$z = r \sin \phi$$

$$\dot{x} = v (\sin \gamma \cos \phi \cos \theta - \cos \gamma \sin \sigma \sin \theta - \cos \gamma \cos \sigma \sin \phi \cos \theta)$$

$$\dot{y} = v (\sin \gamma \cos \phi \sin \theta + \cos \gamma \sin \sigma \cos \theta - \cos \gamma \cos \sigma \sin \phi \sin \theta)$$

$$\dot{z} = v (\sin \gamma \sin \phi + \cos \gamma \cos \sigma \cos \phi)$$

The definitions of pertinent quantities are apparent in the following figure.



SUBROUTINE COPINS

PURPOSE: TO DETERMINE THE IMPULSIVE CORRECTION AND TIME REQUIRED TO INSERT FROM AN APPROACH HYPERBOLA INTO A COPLANAR ELIPTICAL ORBIT.

CALLING SEQUENCE: CALL COPINS(GM,R,V,DA,DE,DELW,TEX,DELV,IEX)

ARGUMENTS:

GM	I	GRAVITATIONAL CONSTANT
R(3)	I	POSITION VECTOR AT DECISION
V(3)	I	VELOCITY VECTOR AT DECISION
DE	I	DESIRED SEMIMAJOR AXIS
DE	I	DESIRED ECCENTRICITY
DELW	I	DESIRED PERIAPSIS SHIFT
TEX	O	TIME FROM DECISION TO EXECUTION (SECONDS)
DELV(3)	O	INSERTION VELOCITY CORRECTION
IEX	O	EXECUTION CODE =0 EVENT IS EXECUTABLE =1 NO EXECUTABLE SOLUTION FOUND

SUBROUTINES SUPPORTED: INSERS

SUBROUTINES REQUIRED: CAREL ELCAR

LOCAL SYMBOLS:

AA	COEFFICIENT DEFINING TANGENTIAL SOLUTION FOR A
AH	HYPERBOLIC SEMIMAJOR AXIS
ARC	THE CONSTANT 180
A1	CANIDATE SOLUTION FOR SEMIMAJOR AXIS
A2	CANIDATE SOLUTION FOR SEMIMAJOR AXIS
A	TARGET SEMIMAJOR AXIS
BB	COEFFICIENT DEFINING TANGENTIAL SO[illegible]ION FOR A
B	TANGENTIAL SOLUTION CONSTANT
CC	COEFFICIENT DEFINING TANGENTIAL SOLUTION FOR A

COE	1/E
COSTH	COS(THETA)
COSW	COS(W)
C	TANGENTIAL SOLUTION CONSTANT
DELVM	MAGNITUDE OF FINAL CORRECTION
DISC	DISCRIMINANT OF SOLUTION FOR THETA
DISK	DISCRIMINANT OF TANGENTIAL SOLUTION FOR A
DRA	DESIRED APOAPSIS RADIUS
DRP	DESIRED PERIAPSIS RADIUS
DVM	MAGNITUDE OF VELOCITY CORRECTION FOR CANDIDATE SOLUTION
DV	VELOCITY CORRECTION OF CANDIDATE SOLUTION
D	TANGENTIAL SOLUTION CONSTANT
EH	HYPERBOLIC ECCENTRICITY
ERRMAX	SCALAR ERROR ASSOCIATED WITH IMPOSSIBLE SOLUTION
ERR	SCALAR ERRORS OF CANDIDATE SOLUTIONS
ER	RADIUS ON ELLIPSE AT INSERTION
ETA	TRUE ANOMALY ON ELLIPSE AT INSERTION
E	ECCENTRICITY OF ELLIPSE
HI	INCLINATION OF HYPERBOLA
HN	ASCENDING NODE OF HYPERBOLA
HRP	HYPERBOLIC PERIAPSIS RADIUS
HR	RADIUS OF HYPERBOLA AT INSERTION
IOPT	TYPE OF SOLUTION =0 ORBITS INTERSECT =1 MUST MODIFY ORBIT TO OBTAIN SOLUTION
ISOL	INDEX OF SOLUTION

MIN	INDEX OF MINIMUM LOSS FUNCTION SOLUTION
NSOLS	NUMBER OF SOLUTIONS
PH	HYPERBOLIC SEMILATUS RECTUM
PI	THE MATHEMATICAL CONSTANT PI
PP	UNIT VECTOR TOWARD PERIAPSIS
P	ELLIPTICAL SEMILATUS RECTUM
QQ	UNIT VECTOR IN ORBIT PLANE NORMAL TO PP
RAD	DEGREE TO RADIAN TRANSFORMATION
RA	APOAPSIS RADIUS
RD	RADIUS TO DECISION STATE
REMG	MAGNITUDE OF RADIUS ON ELLIPSE AFTER INSERTION
RE	POSITION VECTOR ON ELLIPSE AFTER INSERTION
RH	POSITION ON HYPERBOLA BEFORE INSERTION
RMAG	MAGNITUDE OF RADIUS ON HYPERBOLA BEFORE INSERTION
RP	PERIAPSIS RADIUS
SGN	PARAMETER IN TANGENTIAL SOLUTION
SINW	SIN(W)
STA	TRUE ANOMALY ON HYPERBOLA AT DECISION
SYGN	POSITIVE OR NEGATIVE SIGN IN QUADRATIC
S	INTERMEDIATE VARIABLE
TFPE	TIME FROM PERIAPSIS ON ELLIPSE
TFPH	HYPERBOLIC TIME FROM PERIAPSIS AT INSERT
THA	TRUE ANOMALY OF INSERTION ON HYPERBOLA
TIMDX	TIME FROM DECISION TO EXECUTION
TIMD	TIME FROM PERIAPSIS AT DECISION

VD	SPEED [illegible] DECISION
VEMG	SPEED ON ELLIPSE AFTER INSERTION
VE	VELOCITY VECTOR ON ELLIPSE AFTER INSERTION
VH	[illegible] ECTOR ON HYPERBLOA BEFORE [illegible]
[illegible]	[illegible] HYPERBOLA BEFORE INSERTION
[illegible]	[illegible] ARGUMENT OF PERIAPSIS
[illegible]	[illegible] TIAL PLANE
	[illegible] ELLIPSE
[illegible]	[illegible] DEFINING HYPERBOLIC [illegible]
[illegible]	[illegible] DEFINING HYPERBOLIC [illegible] ATION
	[illegible] DEFINING HYPERBOLIC [illegible] RTION

COPINS Analysis:

COPINS determines the impulsive correction and time required to insert from an approach hyperbola into a coplanar elliptical orbit. The approach hyperbola is specified by a planetocentric state $\vec{r}, \vec{v}$ at a decision time t_d. The desired elliptical orbit is prescribed by input parameters a, e, $\Delta\omega$ where a and e are the semi-major axis and eccentricity of the desired ellipse and $\Delta\omega$ is the angle (measured counter clockwise) from the hyperbolic periapsis to the periapsis of the desired orbit. The situation is illustrated in Figure 1.

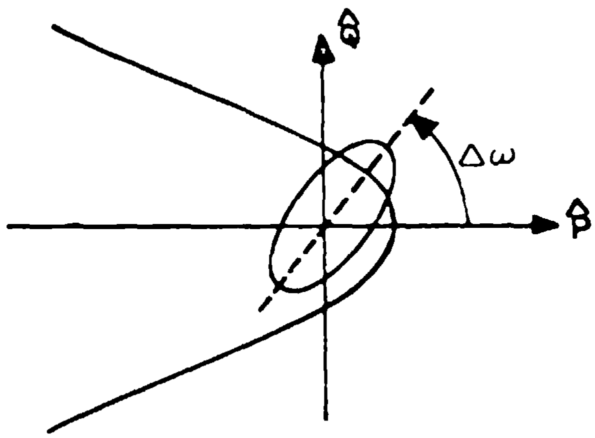


Figure 1. Approach Hyperbola and Desired Orbit

The planetocentric ecliptic state $\vec{r}, \vec{v}$ at the time of decision t_d is first converted to Keplerian elements $(a_H, e_H, i_H, \ _H, \Omega_H, t_{Hd})$ via subroutine CAREL where t_{Hd} is the time from periapsis (negative on the approach ray). The angle f_∞ between the hyperbolic periapsis and the approach asymptote $\hat{S}$ is computed from

$$\cos f_\infty = \frac{1}{e} \qquad 0 < f_\infty < 90^\circ \tag{1}$$

Thus the angle ω between the hyperbolic periapsis and the desired elliptical periapsis is given by

$$\omega = \Delta\omega \tag{2}$$

The hyperbola and ellipse may therefore be described in the PQ plane by standard conic formula, specifically,

$$r_H = \frac{p_H}{1 + e_H \cos\theta}$$
$$r_E = \frac{p_E}{1 + e_E \cos(\theta - \omega)} \qquad (3)$$

where θ is measured counter-clockwise from $\hat{P}$ and p_H, p_E are the semi-latus rectum of the hyperbola and ellipse respectively. Obviously if an angle of intersection θ^* is known, the states on both conics ($\vec{r}^*$, $\vec{v}_H^*$) and ($\vec{r}^*$, $\vec{v}_E^*$) may be computed from conic formulae and the desired impulsive correction is given by

$$\vec{\Delta v} = \vec{v}_E^* - \vec{v}_H^* \qquad (4)$$

Likewise the time from periapsis to the intersection point t^* may be computed using hyperbolic formula and therefore the time from decision to execution is given by

$$\Delta t = t^* - t_d \qquad (5)$$

Thus the coplanar insertion problem reduces to the determination of the optimal angle θ^* for the impulsive maneuver.

From (3) the values of θ for which $r_H = r_E$ are given by

$$\cos\theta = \frac{-xy \pm z\sqrt{D}}{y^2 + z^2} \qquad (6)$$

where

$$\begin{aligned} x &= p_H - p_E \\ y &= p_H e_E \cos\omega - p_E e_H \\ z &= p_H e_E \sin\omega \\ D &= y^2 + z^2 - x^2 \end{aligned} \qquad (7)$$

If the discriminant $D \geq 0$ there are at most two real non-extraneous solutions θ_1, θ_2 such that $r_E(\theta) = r_H(\theta)$. Note that the angle θ may not lie in the region inside the approach and departure asymptotes. If there are two solutions, both Δv 's are computed by (4) and the minimum Δv transfer is selected.

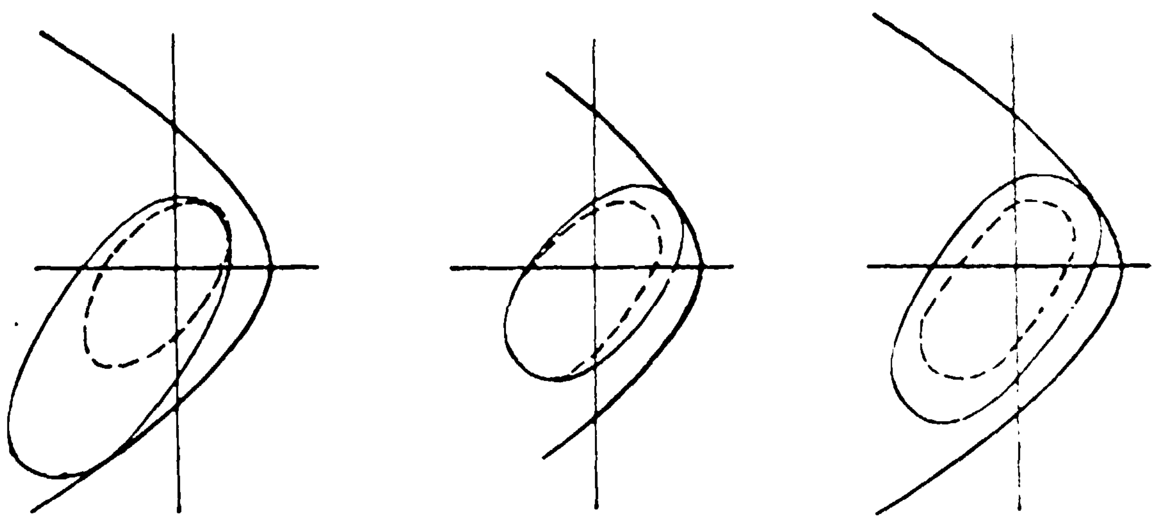
If $D < 0$, the applied hyperbola and the desired orbit do not intersect and there is no impulsive transfer between the two conics. In such a case the desired elements a_E and e_E are modified to determine the "best" tangential solution possible. Three different modifications are tested:

(1) Vary r_a while holding r_p at the desired value.

(2) Vary r_p while holding r_a at the desired value.

(3) Vary a_E while holding e_E at the desired value.

The three modification schemes are illustrated in Figure 2 where the original nonintersecting orbit is shown by the broken lines.



a. Modify r_a b. Modify r_p c. Modify a

Figure 2. Candidate Orbit Modifications

It is desired to modify the "a" and the "e" of the desired orbit to achieve the tangential configurations. From (6) it is obvious that a necessary condition for a tangential solution is given by D=0. Using (7) D may be written

$$D = p_H^2 (e_E^2 - 1) + p_E^2 b + 2p_H p_E - cp_E e_E$$

where

$$b = e_H^2 - 1$$

$$c = 2p_H e_H \cos\omega \qquad (8)$$

where it is observed the approach hyperbola is fixed and it is desired not to vary the ω of the desired ellipse so that subsequent apsidal rotations are avoided.

Modification Option 1: Rewriting (8a) in terms of a and r_p leads to

$$a^2 D = (4\,r_p^2\,b + 4\,r_p\,p_H - 2\,r_p\,c)\,a^2 + (-2\,p_H^2\,r_p - 4\,r_p^3\,b - 2\,p_H\,r_p^2 + 3\,r_p^2\,b)a + (p_H^2\,r_p^2 + r_p^4\,b - c\,r_p^3) \quad (9)$$

Now if D is set equal to 0, r_p held at its desired value, and the resulting quadratic solved for "a", the solution will correspond to the tangential solution which holds r_p constant. If $a \leq 0$ or imaginary, the solution is disregarded. The modified eccentricity is of course defined by

$$e = 1 - \frac{r_p}{a} \quad (10)$$

Modification Option 2: Rewriting (8a) in terms of a and r_a leads to

$$a^2 D = (4\,r_a^2\,b + 4\,r_a\,p_H + 2\,r_a\,c)a^2 + (-2\,p_H^2\,r \; - 4\,r_a^3\,b - 2\,p_H\,r_a^2 - 3\,r_a^2\,c)a + (p_H^2\,r_a^2 + r_a^4\,b + c\,r_a^3) \quad (11)$$

For computational purposes the similarity between (9) and (11) may be exploited. Again setting $D = 0$ and holding r_a at its desired value, the value of "a" may be determined which specifies the tangential solution holding r_a constant. Having determined a realistic value of "a", the corresponding eccentricity is given by

$$e = \frac{r_a}{a} - 1 \quad (12)$$

Modification Option 3: Rewriting (8a) in terms of a and e_E leads to

$$D = (d^2 b)a^2 + (2p_H d - cd\,e_E)a - dp_H^2$$
$$d = (1 - e_E^2) \quad (13)$$

Setting $D = 0$ and solving for "a" while holding e_E at its desired value then defines the option 3 solution.

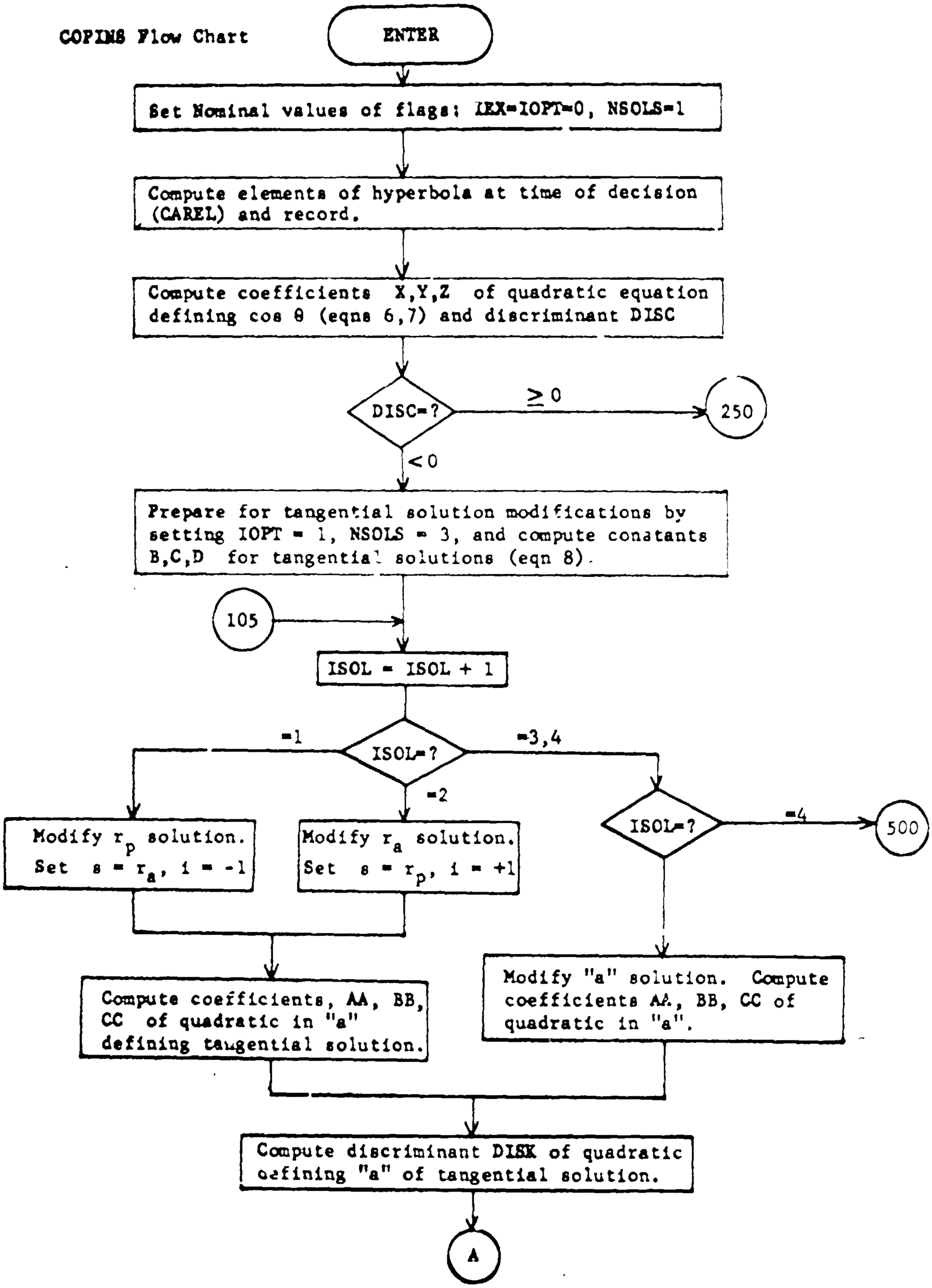
To determine the "best" modified orbit from the three candidate options a rather arbitrary scheme is used. A scalar error is assigned to each option according to a weighting factor and the difference between the desired and achieved values of the periapsis and apoapsis radii:

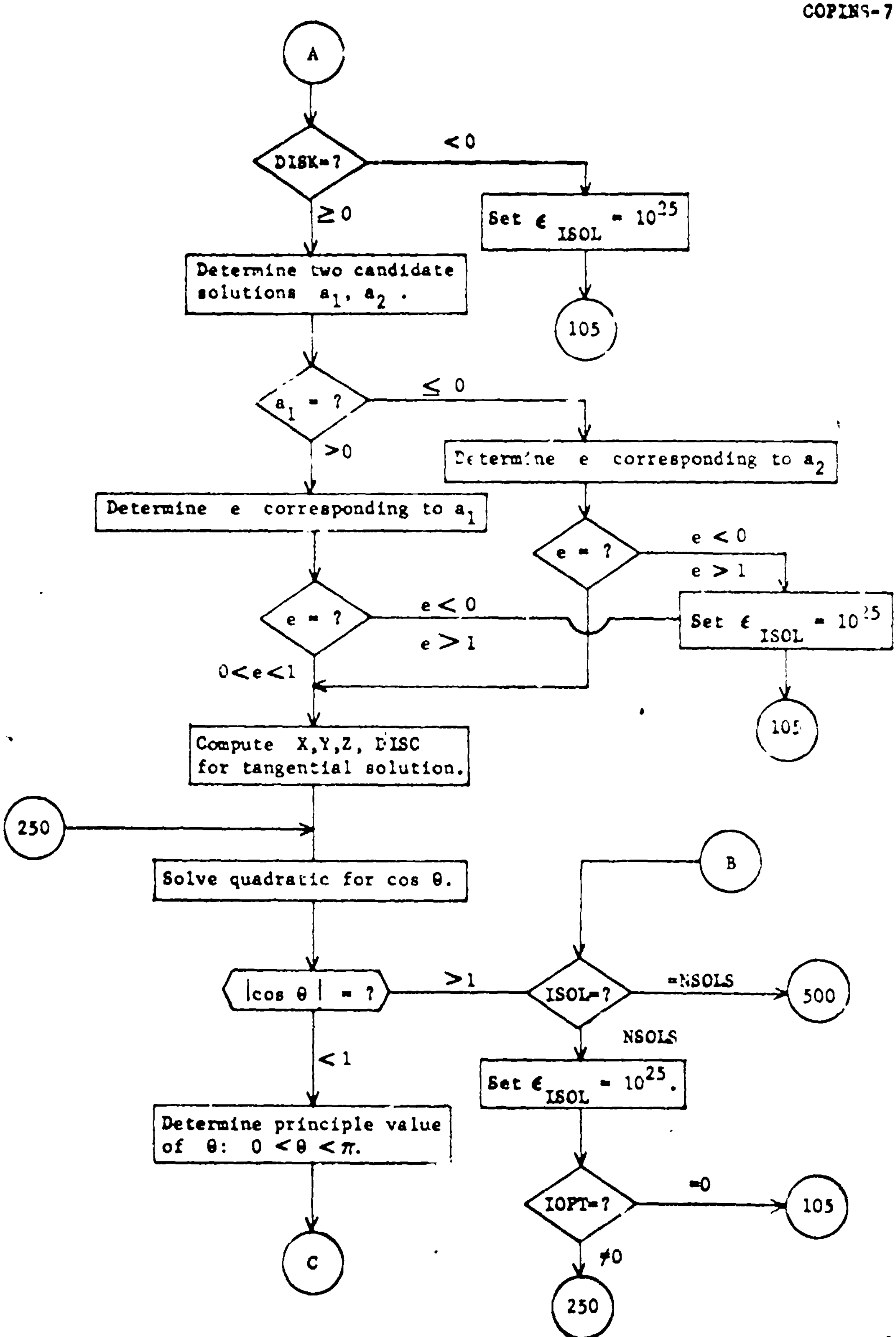
$$E_i = W_i \left(\left| \Delta r_a \right| + \left| \Delta r_p \right| \right)$$

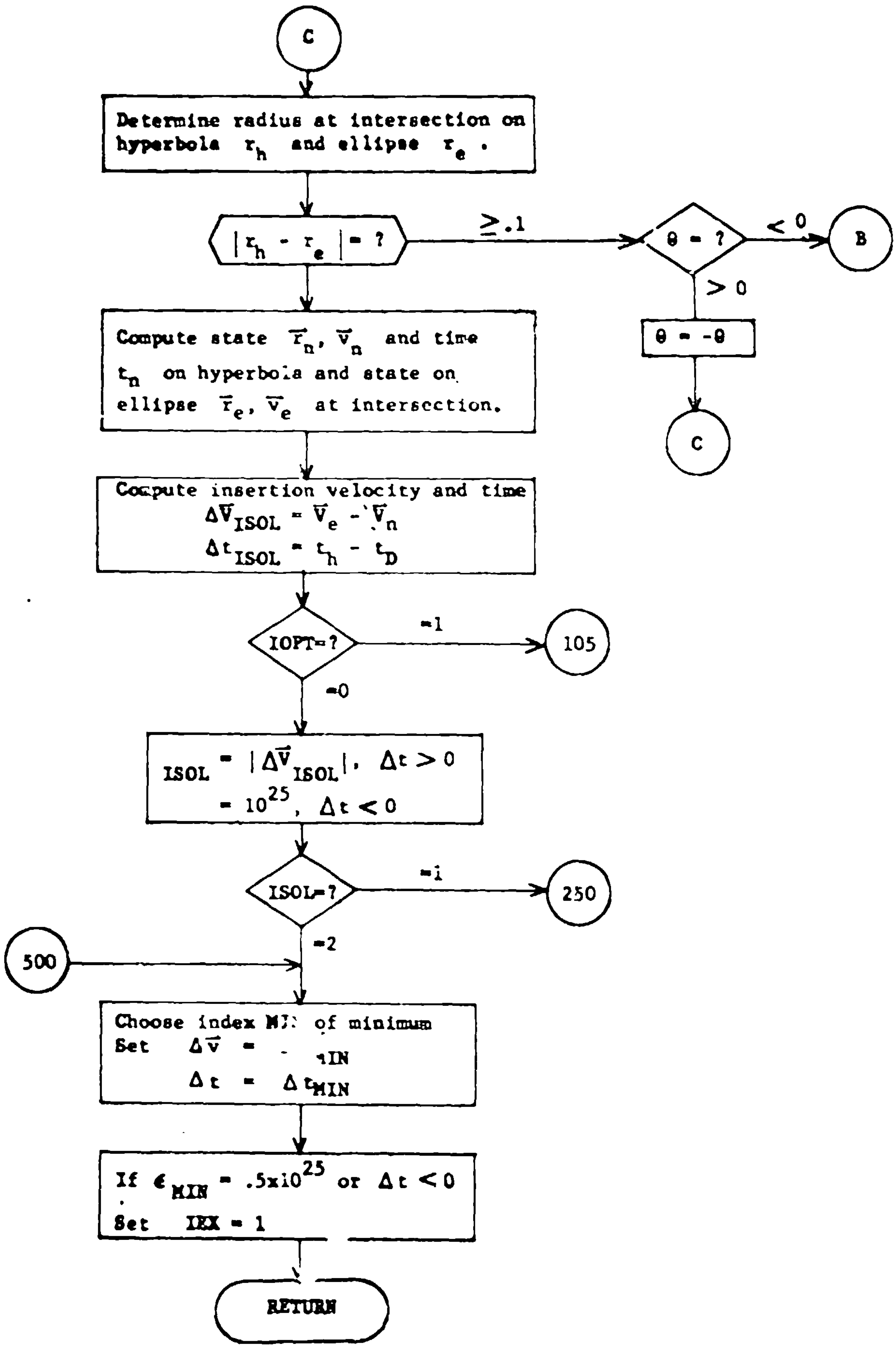
where the scalar factor W_i is set to 1,2,3 respectively for the three [illegible] Thus the preferred strategy is the one which requires a correction only at apoapsis while the least desired scheme requires subsequent corrections [illegible] periapsis and apoapsis.

[illegible] determined orbital elements that necessarily lead to a tangential [illegible] (o) may now be used to compute the angle of intersection θ .

COPINS Flow Chart




A
DISK = ?
< 0
≥ 0
Set ε ISOL = 10^25
105
Determine two candidate solutions a1, a2 .
a1 = ?
≤ 0
> 0
Determine e corresponding to a2
Determine e corresponding to a1
e = ?
e < 0
e > 1
Set ε ISOL = 10^25
e = ?
e < 0
e > 1
0<e<1
105
Compute X,Y,Z, DISC for tangential solution.
250
Solve quadratic for cos θ.
B
|cos θ| = ?
> 1
ISOL = ?
=NSOLS
500
NSOLS
< 1
Set ε ISOL = 10^25.
Determine principle value of θ: 0 < θ < π.
IOPT = ?
=0
105
≠0
C
250


C
Determine radius at intersection on hyperbola r_h and ellipse r_e .
$|r_h - r_e| = ?$
$\geq .1$
$\theta = ?$
< 0
B
> 0
$\theta = -\theta$
C
Compute state $\vec{r}_n$, $\vec{v}_n$ and time t_n on hyperbola and state on ellipse $\vec{r}_e$, $\vec{v}_e$ at intersection.
Compute insertion velocity and time
$\Delta\vec{V}_{ISOL} = \vec{V}_e - \vec{V}_n$
$\Delta t_{ISOL} = t_h - t_D$
IOPT=?
=1
105
=0
$ISOL = |\Delta\vec{V}_{ISOL}|, \ \Delta t > 0$
$= 10^{25}, \ \Delta t < 0$
ISOL=?
=1
250
=2
500
Choose index of minimum
Set $\Delta\vec{v}$ =
$\Delta t = \Delta t_{MIN}$
If $\epsilon_{MIN} = .5x10^{25}$ or $\Delta t < 0$
Set IEX = 1
RETURN

SUBROUTINE CORREL

PURPOSE: CONVERT COVARIANCE MATRIX PARTITIONS TO CORRELATION MATRIX PARTITIONS AND STANDARD DEVIATIONS AND WRITE THEM OUT

CALLING SEQUENCE: CALL CORREL(PP,CXXSP,PSP,CXUP,U0,CXVP,V0,CXSUP,CXSVP)

ARGUMENT:

PP	I	POSITION/VELOCITY COVARIANCE MATRIX
CXXSP	I	CORRELATION BETWEEN SOLVE-FOR PARAMETERS AND POSITION/VELOCITY STATE
PSP	I	SOLVE-FOR PARAMETER COVARIANCE MATRIX
CXUP	I	CORRELATION BETWEEN POSITION/VELOCITY STATE AND DYNAMIC CONSIDER PARAMETERS
U0	I	DYNAMIC CONSIDER PARAMETER COVARIANCE MATRIX
CXVP	I	CORRELATION BETWEEN POSITION/VELOCITY STATE AND MEASUREMENT CONSIDER PARAMETERS
V0	I	MEASUREMENT CONSIDER PARAMETER COVARIANCE MATRIX
CXSUP	I	CORRELATION BETWEEN SOLVE-FOR PARAMETERS AND DYNAMIC CONSIDER PARAMETERS
CXSVP	I	CORRELATION BETWEEN SOLVE-FOR PARAMETERS AND MEASUREMENT CONSIDER PARAMETERS

SUBROUTINES SUPPORTED: PRINT4 SETEVS GUISIM PRESIM PRNTS4 PRINT3 SETEVN GUIDM PRED PRNTS3

LOCAL SYMBOLS:

DUM	INVERSE OF SQUARE ROOT OF DIAGONAL ELEMENTS IN DYNAMIC AND MEASUREMENT CONSIDER COVARIANCE PARTITIONS
IEND	COUNTER INDICATING TOTAL NUMBER OF AUGMENTED STATE VARIABLES
ROW	INTERMEDIATE COMPUTATION AND OUTPUT VECTOR
SQP	INVERSE OF THE SQUARE ROOT OF DIAGONAL ELEMENTS IN VEHICLE AND SOLVE-FOR COVARIANCE PARTITIONS
ZZ	STANDARD DEVIATION

COMMON USED: KPRINT NDIM1 NDIM2 NDIM3 ONE

XLAB XSL XU XV

SUBROUTINE DATA

PURPOSE: TO READ INPUT DATA, TRANSLATE THIS DATA INTO PROPER INTERNAL VALUES, ASSIGN VALUES TO UNSPECIFIED NAMELIST VARIABLES, SET NECESSARY INITIAL VALUES, COMPUTE DIMENSIONS OF STATE TRANSITION, OBSERVATION, AND COVARIANCE MATRIX PARTITIONS, ORDER MEASURMENT AND EVENT SCHEDULES, AND PRINT OUT INITIAL CONDITIONS IN THE ERRAN PROGRAM.

CALLING SEQUENCE: CALL DATA

ARGUMENT: NONE

SUBROUTINES SUPPORTED: ERRON

SUBROUTINES REQUIRED: CONVRT EPHEM GHA ORB PECEQ TIME TRANS XYZRV

LOCAL SYMBOLS:

AI	INCLINATION
AMIN	INTERMEDIATE VARIABLE
ANODE	LONGITUDE OF ASCENDING NODE
A	SEMIMAJOR AXIS
DUM1	INTERMEDIATE STOR . ARRAY
DUM	INTERMEDIATE STORAGE ARRAY
D	INTERMEDIATE JULIAN DATE
DATE	ARRAY CONTAINING FINAL JULIAN DATE
EARTH	CALENDAR DATE AT WHICH EARTHS ORBITAL ELEMENTS WILL BE CALCULATED
E	ECCENTRICITY
FNDT	DATE OF FINAL TIME
GAMMA	PATH ANGLE
GM	GRAVITATIONAL CONSTANT OF CENTRAL BODY
ICNT	COUNTER
IDAY	CALENDAR DAY OF FINAL TIME

IHR	CALENDAR HOUR OF FINAL TIME
IMIN	CALENDAR MINUTES OF FINAL TIME
IMO	CALENDAR MONTH OF FINAL TIME
IYR	CALENDAR YEAR OF FINAL TIME
JUPITER	CALENDAR DATE AT WHICH ORBITAL ELEMENTS OF JUPITER WILL BE CALCULATED
LDAY	CALENDAR DAY OF INITIAL TIME
LHR	CALENDAR HOURS OF INITIAL TIME
LMIN	CALENDAR MINUTES OF INITIAL TIME
LMO	CALENDAR MONTH OF INITIAL TIME
LYR	CALENDAR YEAR OF INITIAL TIME
MARS	CALENDAR DATE AT WHICH ORBITAL ELEMENTS OF MARS WILL BE CALCULATED
MERCURY	CALENDAR DATE AT WHICH ORBITAL ELEMENTS OF MERCURY WILL BE CALCULATED
MOON	CALENDAR DATE AT WHICH ORBITAL ELEMENTS OF EARTHS MOON WILL BE CALCULATED
NENT	NUMBER OF ENTRIES IN MEASUREMENT SCHEDULE
NEPTUNE	CALENDAR DATE AT WHICH ORBITAL ELEMENTS OF NEPTUNE WILL BE CALCULATED
OME	ARGUMENT OF PERIAPSIS
PHIT	DECLINATION
PLUTO	CALENDAR DATE AT WHICH ORBITAL ELEMENTS OF PLUTO WILL BE CALUCLATED
PRD	INITIAL SEMI-LATUS RECTUM OF SPACECRAFT ORBIT
RDS	GEOCENTRIC RADIUS OF VEHICLE
RD	MAGNITUDE OF POSITION VECTOR
SATURN	CALENDAR DATE AT WHICH ORBITAL ELEMENTS OF SATURN WILL BE CALCULATED
SEC	INTERMEDIATE CALENDAR SECONDS

SECI	CALENDAR SECONDS AT FINAL TIME
SECL	CALENDAR SECONDS AT INITIAL TIME
SIGMA	AZIMUTH
TA	TRUE ANOMALY OF INSTANTANEOUS POSITION AND VELOCITY
THETA	RIGHT ASCENSION
URANUS	CALENDAR DATE AT WHICH ORBITAL ELEMENTS OF URANUS WILL BE CALCULATED
VEL	INJECTION VELOCITY RELATIVE TO EARTH
VENUS	CALENDAR DATE AT WHICH ORBITAL ELEMENTS OF VENUS WILL BE CALCULATED
VL	MAGNITUDE OF VELOCITY VECTOR
VUNIT	INTERMEDIATE VELOCITY CONVERSION FACTOR

COMMON COMPUTED/USED:

ACCND	ACC	ALNGTH	CXSU	CXSV
CXU	CXV	CXXS	DELAXS	DELECC
DELICL	DELMA	DELMUP	DELMUS	DELNOD
DELTP	DELW	DTMAX	DTPLAN	DTSUN
EM1	EM4	EM5	EM6	EM7
EM8	EPS	EP50	FACP	FACV
FNTM	FOP	FOV	IAUGIN	IBARY
ICDQ3	ICDT3	ICOORD	ICOOR	IDNF
IEIG	IEPHEM	IEVNT	IHYP1	IMNF
INPR	IOPT7	IPRINT	IPRT	ISP2
ISTMC	ISTM1	KPRINT	MNCN	NBOD
NDACC	NDIM1	NDIM2	NDIM3	NEV10
NEV11	NEV1	NEV2	NEV3	NEV4
NEV5	NEV6	NEV7	NEV9	NEV
NMN	NO	NST	NTMC	ONE
PS	P	RAD	SAL	SIGALP
SIGBET	SIGPRO	SIGRES	SLAT	SLON
SSS	TEV	TMN	TM	TRTM1
TMO	T	UST	U0	VST
V0	WST	XI	XV	ZERO

COMMON COMPUTED:

BDRSI1	BDRSI2	BDRSI3	BDTSI1	BDTSI2
BDTSI3	BSI1	BSI2	BSI3	CXSUB
CXSUG	CXSVB	CXSVG	CXUB	CXUG
CXVB	CXVG	CXXSB	CXXSG	DELTM
EM13	EM2	EM3	EM9	EM
HALF	IAUGDC	IAUGMC	IAUG	ICA1
ICA2	ICA3	ICL2	ICL	IIPOL

INCMT	INITAL	IPOL	ISOI1	ISOI2
ISOI3	ISPH	ITR	MCNTR	MCODE
NAE	NBODYI	NEV8	NGE	NPE
NQE	OMEGA	PB	PG	PSB
PSG	RCA1	RCA2	RCA3	RSOI1
RSOI2	RSOI3	TCA1	TCA2	TCA3
TG	THREE	TIMINT	TRTMB	TSOI1
TSOI2	TSOI3	TWOPI	VSOI1	VSOI2
VSOI3	XB	XF	XG	XSL
XJ	XV			

COMMON USED:

AINC7	ANODE7	DATEJ	DNCN	ECC7
ELMNT	EVNM	F	G	HP7
IPROB	MNNAME	NB	NLP	NTP
PERP7	PI	PLANET	PMASS	P7
TAU7	TPT2	XLAB	XNM	

PROGRAM DATAS

PURPOSE: TO READ INPUT DATA, TRANSLATE THIS DATA INTO PROPER INTERNAL VALUES, ASSIGN VALUES TO UNSPECIFIED NAMELIST VARIABLES, SET NECESSARY INITIAL VALUES, COMPUTE DIMENSIONS OF STATE TRANSITION, OBSERVATION, AND COVARIANCE MATRIX PARTITIONS, ORDER MEASURMENT AND EVENT SCHEDULES, AND PRINT OUT INITIAL CONDITIONS IN THE SIMUL PROGRAM.

CALLING SEQUENCE: CALL DATAS

SUBROUTINES SUPPORTED: MAIN

SUBROUTINES REQUIRED: CONVRT DATA1S ELCAR EPHEM ORB PECEQ TIME TRANS

LOCAL SYMBOLS

AI	INITIAL INCLINATION OF SPACECRAFT ORBIT
ANODE	INITIAL LONGITUDE OF ASCENDING NODE OF SPACECRAFT ORBIT
A	INITIAL SEMI-MAJOR AXIS OF SPACECRAFT ORBIT
DATE	ARRAY CONTAINING FINAL JULIAN DATE
DUM1	PLANETO-CENTRIC ECLIPTIC SPACECRAFT STATE
DUM	COORDINATE TRANSFORMATION FROM PLANETO-CENTRIC EQUATORIAL TO PLANETO-CENTRIC ECLIPTIC COORDINATES
D	JULIAN DATE AT LAUNCH
EARTH	CALENDAR DATE AT WHICH ORBITAL ELEMENTS OF EARTH WILL BE CALCULATED
E	INITIAL ECCENTRICITY OF SPACECRAFT ORBIT
FNOT	FINAL JULIAN DATE
GAMMA	INJECTION PATH ANGLE
GM	GRAVITATIONAL CONSTANT OF TARGET PLANET
IDAY	DAY OF FINAL COMPUTATION
IHR	HOUR OF FINAL COMPUTATION
IMIN	MINUTE OF FINAL COMPUTATION
IMO	MONTH OF FINAL COMPUTATION

IYR	YEAR OF FINAL COMPUTATION
JUPITER	CALENDAR DATE AT WHICH ORBITAL ELEMENTS OF JUPITER WILL BE CALCULATED
LDAY	LAUNCH DAY
LHR	LAUNCH HOUR
LMIN	LAUNCH MINUTE
LMO	LAUNCH MONTH
LYR	LAUNCH YEAR
MARS	CALENDAR DATE AT WHICH ORBITAL ELEMENTS OF MARS WILL BE CALCULATED
MERCURY	CALENDAR DATE AT WHICH ORBITAL ELEMENTS OF MERCURY WILL BE CALCULATED
MOON	CALENDAR DATE AT WHICH ORBITAL ELEMENTS OF MOON WILL BE CALCULATED
NENT	NUMBR OF ENTRIES IN MEASUREMENT SCHEDULE
NEPTUNE	CALENDAR DATE AT WHICH ORBITAL ELEMETNS OF NEPTUNE WILL BE CALCULATED
OME	INITIAL ARGUMENT OF PERIAPSIS OF SPACE-CRAFT ORBIT
PHIT	INJECTION DECLINATION
PLUTO	CALENDAR DATE AT WHICH ORBITAL ELEMENTS OF PLUTO WILL BE CALCULATED
PRD	INITIAL SEMI-LATUS RECTUM OF SPACECRAFT ORBIT
RDS	EARTH-CENTERED INJECTION RADIUS
RD	DUMMY VARIABLE
SATURN	CALENDAR DATE AT WHICH ORBITAL ELEMENTS OF SATURN WILL BE CALCULATED
SECI	SECOND OF FINAL COMPUTATION
SECL	LAUNCH SECOND
SEC	SECOND OF CALENDAR DATE AT WHICH ORBITAL

	ELEMENTS OF A PLANET WILL BE CALCULATED
SIGMA	INJECTION AZIMUTH
TA	INITIAL SPACECRAFT TRUE ANOMALY
THETA	INJECTION RIGHT ASCENSION
URANUS	CALENDAR DATE AT WHICH ORBITAL ELEMENTS OF URANUS WILL BE CALCULATED
VEL	INJECTION VELOCITY RELATIVE TO EARTH
VENUS	CALENDAR DATE AT WHICH ORBITAL ELEMENTS OF VENUS WILL BE CALCULATED
VL	DUMMY VARIABLE
VUNIT	VELOCITY CONVERSION FACTOR

COMMON COMPUTED/USED:

AALP	ABET	ACCND	ACC1	ACC
ADEVX	ALNGTH	APRO	ARES	BIA
CXSU	CXSV	CXU	CXV	CXXS
DAB	DEB	DELAXS	DELECC	DELICL
DELMA	DELMUP	DELMUS	DELNOD	DELTP
DELW	DIB	DMAB	DMUPB	DMUSB
DNOB	DTMAX	DTPLAN	DTSUN	DWB
EM1	EM4	EM5	EM6	EM7
EM8	EPS	EP50	FACP	FACV
FNTM	FOP	FOV	IAMNF	IAUGIN
IBARY	ICDT3	ICOORD	ICOOR	IDNF
IEIG	IEPHEM	IHYP1	IMNF	INPR
IOPT7	IPRINT	IPRT	ISP2	ISTMC
ISTM1	KPRINT	MNCN	NBOD1	NBOD
NB1	NDACC	NDIM1	NDIM2	NDIM3
NEV10	NEV11	NEV1	NEV2	NEV3
NEV4	NEV5	NEV6	NEV7	NEV9
NO	NST	NTMC	ONE	PS
P	RAD	SAL	SIGALP	SIGBET
SIGPRO	SIGRES	SLAT	SLB	SLOM
SSS	TM	TRTM1	TTIM1	TTIM2
TWO	T	UNMAC	UST	U0
VST	V0	WST	XI	XP
ZERO				

COMMON COMPUTED:

ADEVXS	BDRSI1	BDRSI2	BDRSI3	BDTSI1
BDTSI2	BDTSI3	BSI1	BSI2	BSI3
DELTM	EM13	EM[illegible]	EM3	EM9
HALF	IAUGDC	IAUGMC	IAUG	ICA1
ICA2	ICA3	IC[illegible]	ICL2	IC[illegible]
INCMT	INITAL	ISOI1	ISOI2	ISOI3
ISPH	ITR	NBODYI	NEV8	RCA1

	RCA2	RCA3	RSOI1	RSOI2	RSOI3
	TCA1	TCA2	TCA3	TEV	THREE
	TIMINT	TRTMB	TSOI1	TSOI2	TSOI3
	TWOPI	VSOI1	VSOI2	VSOI3	XI1
	XSL	XU	XV	ZI	
COMMON USED:	AINC7	ANODE7	AVARM	DATEJ	DNCN
	ECC7	ELMNT	F	HP7	IPROB
	NAF6	NB	NLP	NTP	PERP7
	PI	PLANET	PMASS	P7	TAU7
	TPT2	T1	T2	T3	T4
	T5	T6	T7	XNM	

SUBROUTINE DATA1

PURPOSE: TO CONTINUE THE INITIALIZATION PROCESS DESCRIBED UNDER DATA.

CALLING SEQUENCE: CALL DATA1(NENT)

ARGUMENT: NENT I NUMBER OF CARDS IN THE MEASUREMENT SCHEDULE

SUBROUTINES SUPPORTED: DATA

SUBROUTINES REQUIRED: GHA

LOCAL SYMBOLS:

AMIN	INTERMEDIATE VARIABLE
AP	INTERMEDIATE TIME ARRAY
ICNT	COUNTER ON MEASUREMENT SCHEDULE CARDS
IROW	INTERMEDIATE ROW INDEX
MEAS	MEASUREMENT CODES
NOUT	DIMENSION OF AUGUMENTED COVARIANCE M'TRIX
SCHED	ARRAY OF TIMES IN MEASUREMENT SCHEDULE

COMMON COMPUTED/USED:

IEVNT	NEV	NMN	SLAT	SLON
TEV	TMN	T1	T2	T3
T4	T5	T6	T7	T

COMMON COMPUTED:

CXSUB	CXSUG	CXSVB	CXSVG	CXUB
CXUG	CXVB	CXVG	CXXSB	CXXSG
EM	EPS	IIPOL	IPOL	MCNTR
MCODE	NAE	NGE	NPE	NQE
OMEGA	PB	PG	PSB	PSG
TG	XB	XF	XG	

COMMON USED:

CXSU	CXSV	CXU	CXV	CXXS
DATEJ	DNCN	EM7	EM8	EP50
EVNM	FACP	FACV	FNTM	ICDQ3
IDNF	IGUID	IHYP1	IMNF	ISTMC
MNCN	MNNAME	NDIM1	NDIM2	NDIM3
NEV1	NEV2	NEV3	NEV4	NEV5
NEV6	NEV7	NST	ONE	PS
P	RAD	SAL	TPT2	TRTM1
U0	V0	XI	ZERO	

SUBROUTINE DATA1S

PURPOSE: TO CONTINUE THE INITIALIZATION PROCESS DESCRIBED UNDER DATAS.

CALLING SEQUENCE: CALL DATA1S(NENT)

ARGUMENT: NENT I NUMBER OF CARDS IN THE MEASUREMENT SCHEDULE

SUBROUTINES SUPPORTED: DATAS

SUBROUTINES REQUIRED: GHA

LOCAL SYMBOLS:

AMIN	INTERMEDIATE VARIABLE
AP	INTERMEDIATE TIME ARRAY
ICNT	COUNTER ON MEASUREMENT SCHEDULE CARDS
IROW	INTERMEDIATE ROW INDEX
MEAS	MEASUREMENT CODES
NOUT	DIMENSION OF AUGUMENTED COVARIANCE MATRIX
PARAM	ARRAY OF AUGMENTED BIASES
SCHED	ARRAY OF TIMES IN MEASUREMENT SCHEDULE

COMMON COMPUTED/USED:

ADEVXS	IEVNT	NEV	NMN	SLAT
SLB	SLON	TEV	TMN	T1
T2	T3	T4	T5	T6
T7	T			

COMMON COMPUTED:

ADEVSB	ADEVXB	CXSUB	CXSUG	CXSVB
CXSVG	CXUB	CXUG	CXVB	CXVG
CXXSB	CXXSG	EDEVXS	EDEVX	EM
EPS	IIPOL	IPOL	MCNTR	MCODE
NAE	NGE	NPE	NQE	OMEGA
PB	PG	PSB	PSG	TG
XB	XF	XG	XI1	ZI

COMMON USED:

ACC1	ADEVX	AVARM	BIA	CXSU
CXSV	CXU	CXV	CXXS	DAB
DATEJ	DEB	DIB	DMAB	DMUPB
DMUSB	DNCN	DNOB	DWB	EM7
EM8	EP50	EVNM	FACP	FACV
FNTM	IAMNF	IAUGIN	ICDQ3	ICDT3
IDNF	IHYP1	IMNF	ISTMC	MNCN
MNNAME	NBOD1	NB1	NDIM1	NDIM2
NDIM3	NEV1	NEV2	NEV3	NEV4

NEV5	NEV6	NEV7	NST	ONE
PLANET	PS	P	RAD	SAL
TPT2	TPTM1	TTIM1	TTIM2	UNMAG
U0	V0	XDUM	XI	ZERO

SUBROUTINE DESENT

PURPOSE: TO COMPUTE A CORRECTION TO AN INITIAL VELOCITY BY THE STEEPEST DESCENT OR CONJUGATE GRADIENT TECHNIQUES FOR USE BY TARGET.

CALLING SEQUENCE: CALL DESENT(ERC,IT,KREK,GMP,PP)

ARGUMENTS:

ERC	I	SCALAR ERROR OF CURRENT ITERATE
IT	I/O	ITERATION COUNTER
KREK	I	STEEPEST DESENT RECTIFICATION NUMBER
GMP	I/O	PREVIOUS GRADIENT MAGNITUDE (INPUT) CURRENT GRADIENT MAGNITUDE (OUTPUT)
PP(3)	I/O	PREVIOUS GRADIENT (INPUT) CURRENT GRADIENT (OUTPUT)

SUBROUTINES SUPPORTED: TARGET

SUBROUTINES REQUIRED: TAROPT VMP

LOCAL SYMBOLS:

ACK	CURRENT ACCURACY LEVEL
AER	ABSOLUTE ERRORS OF TARGET PARAMETERS
AUXN	VALUES OF AUXILIARY PARAMETERS OF CURRENT ITERATE
DD	DIRECTIONAL DERIATIVE
DELVM	MAGNITUDE OF PREDICTED CORRECTION
DEVI	DEVIATION OF NOMINALLY-PREDICTED AUXILIARY PARAMETERS FROM CURRENT ITERATE VALUES
DUMM	DUMMY VARIABLES
DUM	DUMMY VARIABLES
DVEE	VELOCITY PERTURBATIONS
DVM	MAXIMUM ALLOWABLE VELOCITY INCREMENT
ERB	SCALAR ERROR OF NOMINALLY-PREDICTED STEP
GC	CURRENT GRADIENT
GMC	MAGNITUDE OF GC
HB	NOMINALLY PREDICTED STEP MAGNITUDE

HH	CORRECTION MAGNITUDE AFTER CONSTRAINTS
HS	CORRECTION MAGNITUDE AFTER PARABOLIC FIT
IEND	FLAG SET TO 1 IF TOLERANCES ACCEPTABLE ON PERTURBED TRAJECTORY
ISP2	SOI STOPPING CONDITION FLAG =0 DO NOT STOP AT SOI =1 STOP AT SOI
PC	DIRECTION OF CORRECTION
PERR	PERTURBED ERRORS
PM	MAGNITUDE OF UNNORMALIZED DIRECTION VECTOR
QC	UNIT VECTOR IN DIRECTION OF GRADIENT
RSF	FINAL STATE OF INTEGRATION

COMMON COMPUTED/USED:	DELTAV	ISPH	RIN	TEN	
COMMON COMPUTED:	ICL2	ICL	INCMT	RRF	
COMMON USED:	AAUX	AC	ATAR	CTOL	DAUX
	DELTAT	DTAR	DVMAX	D1	FAC
	IPHASE	ISTOP	KUR	LEV	LVLS
	NOPAR	PERV	TRTM	TWO	ZERO

DESENT Analysis

DESENT computes a correction to an initial velocity by the steepest descent or conjugate gradient techniques for use by TARGET.

The technique used is determined by the value of METHOD. DESENT takes n steps in the conjugate gradient directions before rectifying by making a steepest descent step where $n = METHOD - 1$. Thus if METHOD = 1, all steps are taken in the steepest descent direction.

Let the current iterate initial state be denoted $\vec{r}, \vec{v}$. Let the scalar error of the auxiliary parameters corresponding to this state be denoted ϵ. Let the perturbation size for the sensitivities be dv.

The current gradient $\vec{g}_c$ is computed by numerical differencing. For the k-th component of $\vec{g}_c$ the corresponding component of velocity is perturbed by dv

$$\vec{v}_p = \vec{v} + dv \left[\delta_{1K}, \delta_{2K}, \delta_{3K}\right]^T \tag{1}$$

The initial state $(\vec{r}, \vec{v}_p)$ is then propagated to the final stopping conditions. Let the auxiliary parameters of that trajectory be denoted $\vec{\alpha}_p$. The error associated with the perturbed state is then

$$\epsilon_p = \vec{W} \cdot (\vec{\alpha}_p - \vec{\alpha}^*) \tag{2}$$

where $\vec{W}$ represents the weighting factors and $\vec{\alpha}^*$ are the desired target conditions. The k-th component of the current gradient is then

$$g_{c_K} = \frac{\epsilon_p - \epsilon}{dv} \tag{3}$$

The corrected gradient is given by

$$\vec{p}_c = \vec{g}_c \qquad \text{steepest descent step}$$

$$= \frac{|g_c|^2}{|g_p|^2} \vec{p}_p + \vec{g}_c \qquad \text{conjugate gradient step} \tag{4}$$

where the subscript c refers to a current parameter, p refers to a previous-step parameter.

The unit vector in the direction of the next step is then given by

$$\vec{q}_c = - \frac{\vec{p}_c}{p_c} \qquad (5)$$

The directional derivative of the scalar error in the the direction $\vec{q}_c$ is

$$d = \vec{g}_c \cdot \vec{q}_c \qquad (6)$$

The nominal step size $\bar{h}$ is computed from a linear approximation to null the error

$$\bar{h} = \frac{\epsilon}{-d} \qquad (7)$$

The initial state corrected by this nominal correction is then propagated to the final stopping conditions and the resulting error $\bar{\epsilon}$ computed. The three conditions

$$\begin{aligned} y(o) &= \epsilon \\ y(\bar{h}) &= \bar{\epsilon} \\ y'(o) &= d \end{aligned} \qquad (8)$$

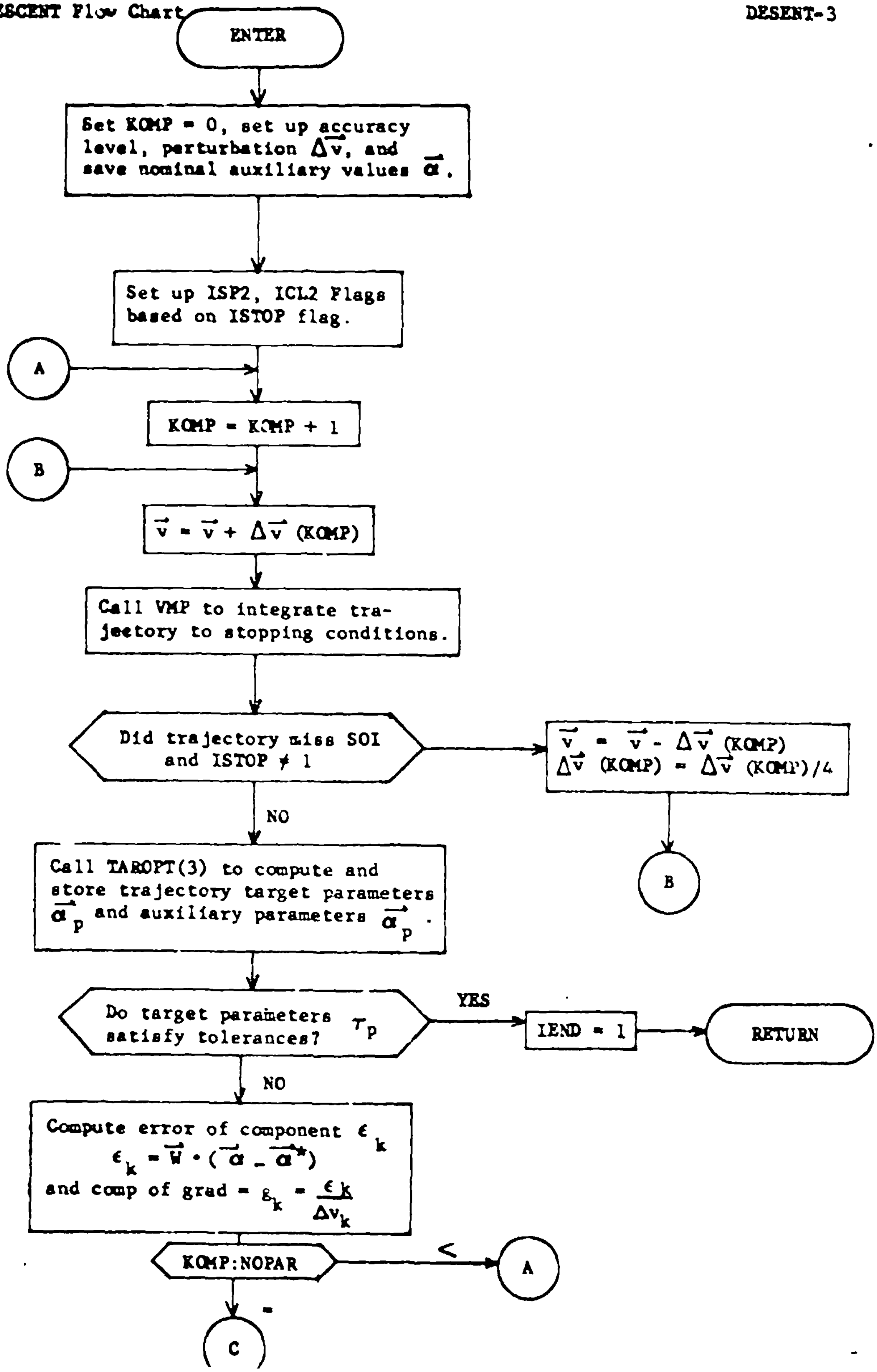
may now be applied to the formula of a parabola $y - \epsilon^* = a(x - h^*)^2$ to predict the optimal step size h^* yielding the minimum error ϵ^*

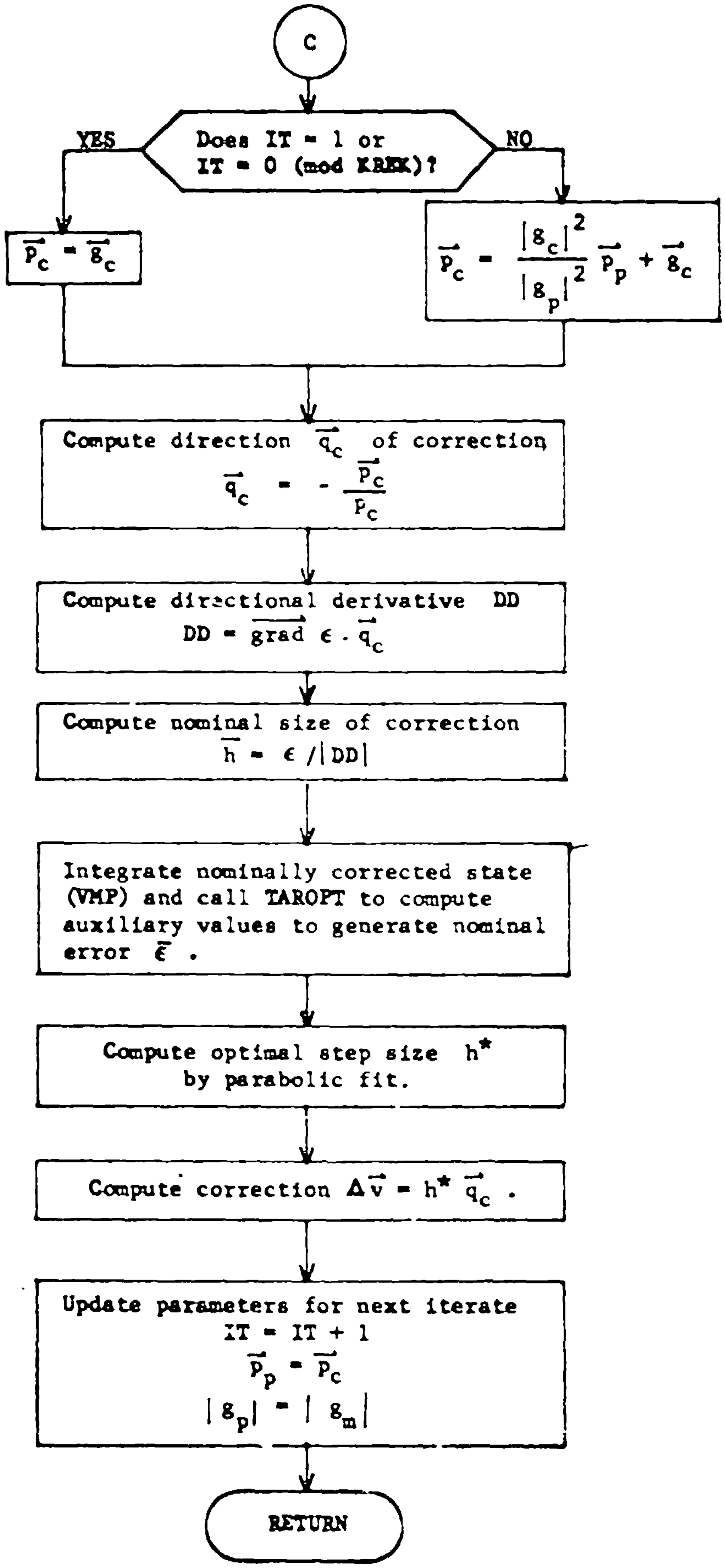
$$h^* = \frac{d\bar{h}^2}{2(d\bar{h} + \epsilon - \bar{\epsilon})} \qquad (9)$$

The correction for the current is then given by

$$\Delta v = h^* \vec{q}_g \qquad (10)$$

Reference: Myers, G. E., "Properties of the Conjugate Gradient and Davidon Methods", AAS Paper 68-081. Presented at 1968 AAS/AIAA Astrodynamics Specialist Conference, Jackson, Wyoming.


ENTER
Set KOMP = 0, set up accuracy level, perturbation $\Delta\vec{v}$, and save nominal auxiliary values $\vec{\alpha}$.
Set up ISP2, ICL2 Flags based on ISTOP flag.
A
KOMP = KOMP + 1
B
$\vec{v} = \vec{v} + \Delta\vec{v}$ (KOMP)
Call VMP to integrate trajectory to stopping conditions.
Did trajectory miss SOI and ISTOP ≠ 1
$\vec{v} = \vec{v} - \Delta\vec{v}$ (KOMP)
$\Delta\vec{v}$ (KOMP) = $\Delta\vec{v}$ (KOMP)/4
B
NO
Call TAROPT(3) to compute and store trajectory target parameters $\vec{\alpha}_p$ and auxiliary parameters $\vec{\alpha}_p$.
Do target parameters satisfy tolerances? τ_p
YES
IEND = 1
RETURN
NO
Compute error of component ϵ_k
$\epsilon_k = \vec{W} \cdot (\vec{\alpha} - \vec{\alpha}^*)$
and comp of grad = $g_k = \frac{\epsilon_k}{\Delta v_k}$
KOMP:NOPAR
<
A
=
C


C
Does IT = 1 or IT = 0 (mod KREK)?
YES
NO
$\vec{p}_c = \vec{g}_c$
$\vec{p}_c = \frac{|g_c|^2}{|g_p|^2}\vec{p}_p + \vec{g}_c$
Compute direction $\vec{q}_c$ of correction
$\vec{q}_c = -\frac{\vec{p}_c}{p_c}$
Compute directional derivative DD
$DD = \overrightarrow{grad}\ \epsilon \cdot \vec{q}_c$
Compute nominal size of correction
$\bar{h} = \epsilon / |DD|$
Integrate nominally corrected state (VMP) and call TAROPT to compute auxiliary values to generate nominal error $\bar{\epsilon}$.
Compute optimal step size h^* by parabolic fit.
Compute correction $\Delta\vec{v} = h^* \vec{q}_c$.
Update parameters for next iterate
$IT = IT + 1$
$\vec{p}_p = \vec{p}_c$
$|g_p| = |g_m|$
RETURN

SUBROUTINE DYNO

PURPOSE: COMPUTE DYNAMIC NOISE COVARIANCE MATRIX IN THE ERROR ANALYSIS PROGRAM

CALLING SEQUENCE: CALL DYNO(ICODE)

ARGUMENT: ICODE I ZERO IN ERROR ANALYSIS MODE

SUBROUTINES SUPPORTED: ERRANN SETEVN GUIDM PRED

LOCAL SYMBOLS: D2 SQUARE OF (DELTM*TM)

COMMON COMPUTED: Q

COMMON USED: DELTM DNCN IDNF TM

DYNØ Analysis

Subroutine DYNØ evaluates the dynamic noise covariance matrix Q over the time interval $\Delta t = t_{k+1} - t_k$. The matrix Q is assumed to have form

$$Q = \mathrm{diag}\left(\frac{1}{4} K_1 \Delta t^4, \frac{1}{4} K_2 \Delta t^4, \frac{1}{4} K_3 \Delta t^4, K_1 \Delta t^2, K_2 \Delta t^2, K_3 \Delta t^2\right)$$

where dynamic noise constants K_1, K_2, and K_3 have units of km^2/sec^4.

SUBROUTINE DYNOS

PURPOSE: COMPUTE DYNAMIC NOISE COVARIANCE MATRIX AND THE ACTUAL DYNAMIC NOISE (UNMODELED ACCELERATION) IN THE SIMULATION PROGRAM

CALLING SEQUENCE: CALL DYNOS(ICODE)

ARGUMENT: ICODE I INTERNAL CODE TO DETERMINE IF THE DYNAMIC NOISE MATRIX IS COMPUTED OR IF THE ACTUAL DYNAMIC NOISE IS CALCULATED

SUBROUTINES SUPPORTED: SIMULL SETEVS GUISIM PRESIM

LOCAL SYMBOLS:

DT	INTERNAL TIME INCREMENT
D2	SQUARE OF (DELTM*TM)
IC	INTERNAL CODE ON DT CALCULATION
T1	CURRENT TIME
T2	CURRENT TIME + DELTA TIME

COMMON COMPUTED/USED: W

COMMON COMPUTED: Q

COMMON USED: DELTM DNCN HALF IDNF TM TRTM1 TTIM1 TTIM2 UNMAC ZERO

DYNØS Analysis

Subroutine DYNØS performs two functions. It's first function is identical to that of subroutine DYNØ, namely, to evaluate the dynamic noise covariance matrix Q over the time interval $\Delta t = t_{k+1} - t_k$.

The second function of subroutine DYNØS is to compute the actual dynamic noise $\vec{\omega}_{k+1}$, which represents the integrated effect of unmodelled accelerations acting on the spacecraft over the time interval Δt. Actual dynamic noise $\vec{\omega}_{k+1}$ is used elsewhere in the program to compute the actual state deviations of the spacecraft from the most recent nominal trajectory.

If we define $\vec{\omega}_{k+1} = \left[\vec{\omega}_{r_{k+1}}, \quad \vec{\omega}_{v_{k+1}} \right]^T$, where

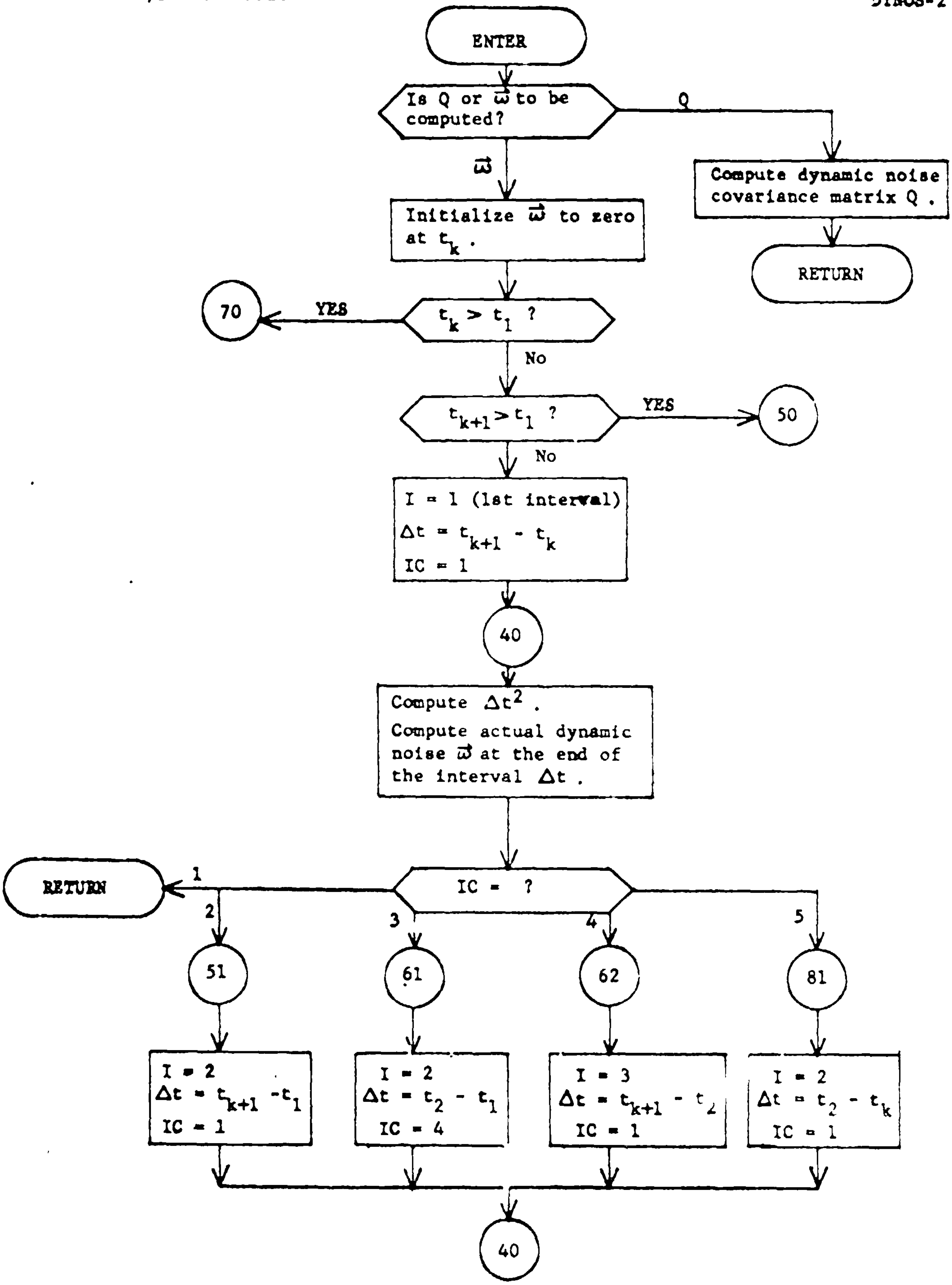
$\vec{\omega}_{r_{k+1}}$ and $\vec{\omega}_{v_{k+1}}$ denote the contributions of unmodelled

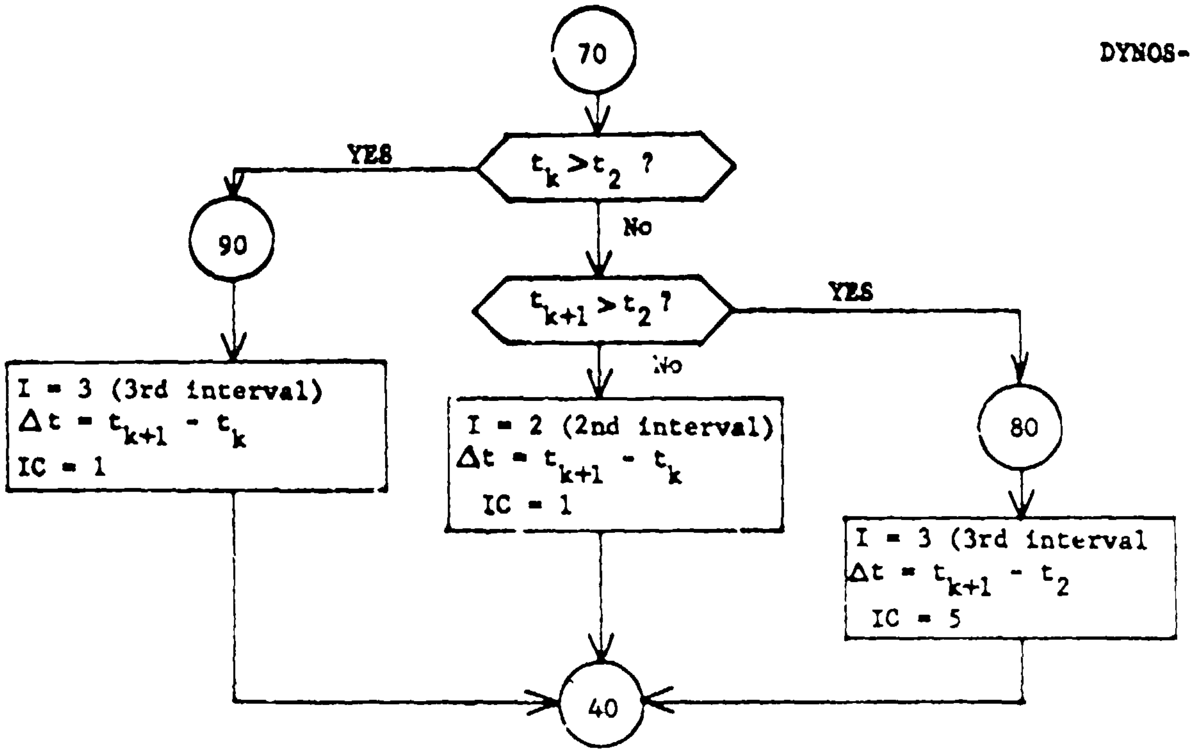
accelerations to spacecraft position and velocity, respectively, and if we assume constant unmodelled acceleration $\vec{a}$, then

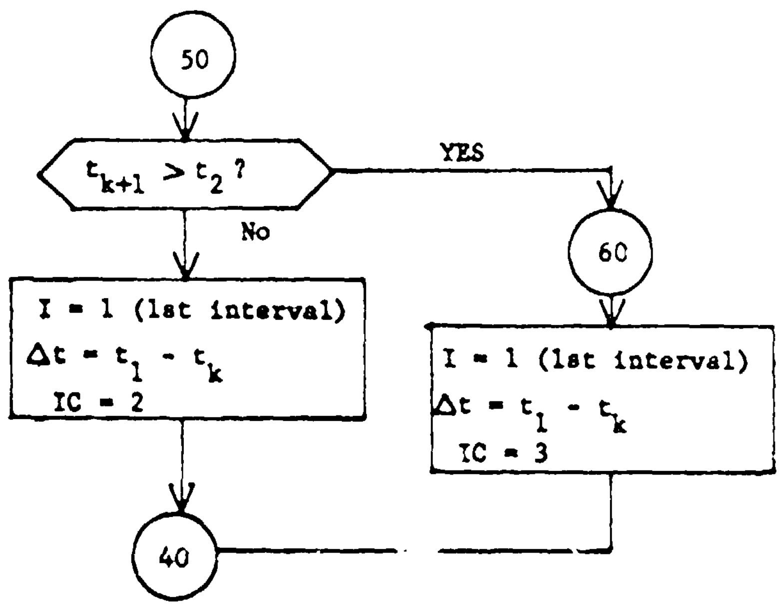
$$\vec{\omega}_{r_{k+1}} = \frac{\vec{a}}{2} (t_{k+1} - t_k)^2 + \vec{\omega}_{v_k} (t_{k+1} - t_k) + \vec{\omega}_{r_k}$$

$$\vec{\omega}_{v_{k+1}} = \vec{a} (t_{k+1} - t_k) + \vec{\omega}_{v_k}$$

The program permits the entire trajectory to be divided into three arbitrary consecutive intervals, over each of which a different constant unmodelled acceleration $\vec{a}$ can be specified. These intervals are represented by (t_o, t_1), (t_1, t_2), and (t_2, t_f), where t_o is the initial trajectory time and t_f is the final trajectory time. If t_k and t_{k+1} occur in different intervals, then the above equations must be evaluated piece-wise over (t_k, t_{k+1}).


ENTER
Is Q or $\vec{\omega}$ to be computed?
Q
Compute dynamic noise covariance matrix Q .
RETURN
$\vec{\omega}$
Initialize $\vec{\omega}$ to zero at t_k .
70
YES
$t_k > t_1$?
No
$t_{k+1} > t_1$?
YES
50
No
I = 1 (1st interval)
$\Delta t = t_{k+1} - t_k$
IC = 1
40
Compute Δt^2 .
Compute actual dynamic noise $\vec{\omega}$ at the end of the interval Δt .
RETURN
1
IC = ?
2
3
4
5
51
61
62
81
I = 2
$\Delta t = t_{k+1} - t_1$
IC = 1
I = 2
$\Delta t = t_2 - t_1$
IC = 4
I = 3
$\Delta t = t_{k+1} - t_2$
IC = 1
I = 2
$\Delta t = t_2 - t_k$
IC = 1
40


70
$t_k > t_2$?
YES
No
90
$t_{k+1} > t_2$?
YES
No
80
I = 3 (3rd interval)
$\Delta t = t_{k+1} - t_k$
IC = 1
I = 2 (2nd interval)
$\Delta t = t_{k+1} - t_k$
IC = 1
I = 3 (3rd interval
$\Delta t = t_{k+1} - t_2$
IC = 5
40


50
$t_{k+1} > t_2$?
YES
No
60
I = 1 (1st interval)
$\Delta t = t_1 - t_k$
IC = 2
I = 1 (1st interval)
$\Delta t = t_1 - t_k$
IC = 3
40

SUBROUTINE EIGHY

PURPOSE: TO CONTROL THE COMPUTATION OF EIGENVALUES, EIGENVECTORS, AND HYPERELLIPSOIDS.

CALLING SEQUENCE: CALL EIGHY(VEIG,FOX,HARG,IFMT)

ARGUMENT:	VEIG	I	MATRIX TO BE DIAGONALIZED
	FOX	I	FINAL OFF-DIAGONAL ANNIHILATION VALUE
	HARG	I	MATRIX FOR WHICH THE HYPERELLIPSOID IS TO BE COMPUTED
	IFMT	I	FORMAT FLAG =1, PRINT POSITION EIGENVALUE TITLE =2, PRINT VELOCITY EIGENVALUE TITLE =3, PRINT EIGENVALUE TITLE

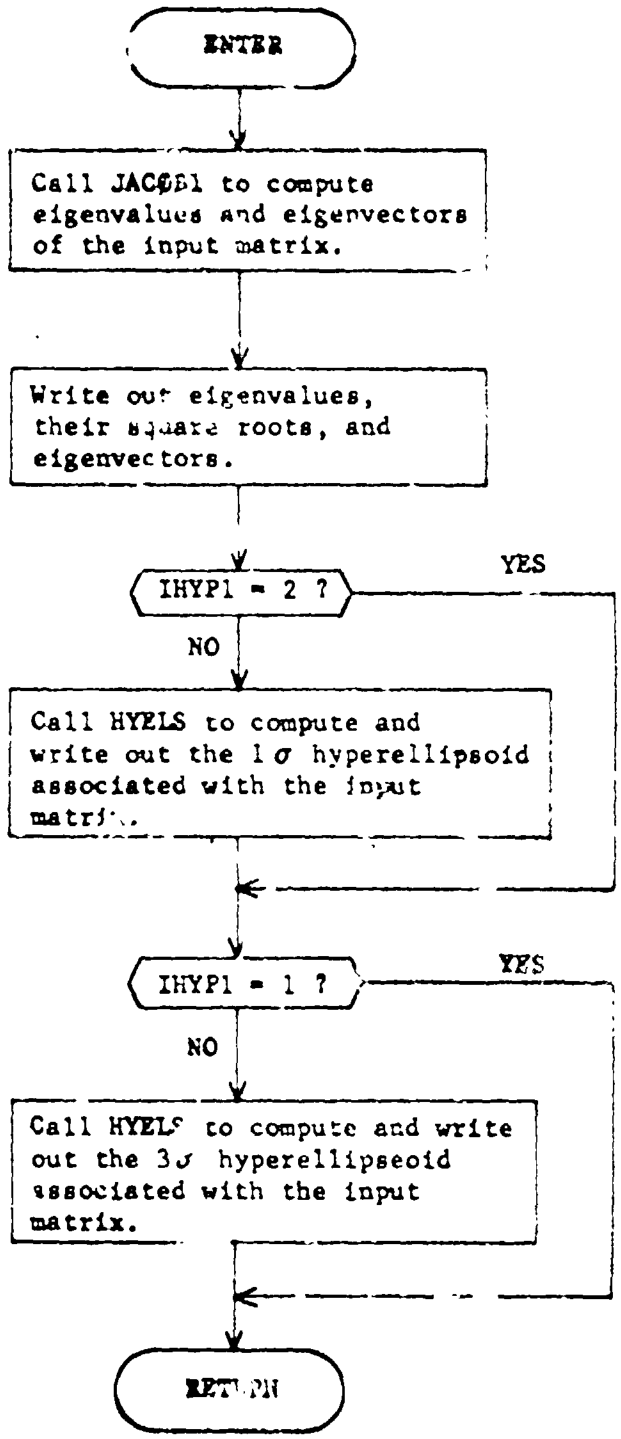
SUBROUTINES SUPPORTED: SETEVS GUISIM GUISS PRESIM GUIDM

SUBROUTINES REQUIRED: HYELS JACOBI

LOCAL SYMBOLS:	EGVCT	EIGENVECTOR MATRIX
	EGVL	EIGENVALUE MATRIX
	OUT	SQUARE ROOTS OF EIGENVALUES

COMMON USED: IHYP1

EIGHY Flow Chart



SUBROUTINE ELCAR

PURPOSE: TRANSFORMATION OF CONIC ELEMENTS TO CARTESIAN COORDINATES

CALLING SEQUENCE: CALL ELCAR(GM,A,E,W,XI,XN,TA,R,RM,V,VM,TFP)

ARGUMENTS

ARGUMENTS	GM	I	GRAVITATIONAL CONSTANT OF CENTRAL BODY
	A	I	SEMIMAJOR AXIS
	E	I	ECCENTRICITY
	W	I	ARGUMENT OF PERIAPSIS
	XI	I	INCLINATION IN REGERENCE SYSTEM
	XN	I	LONGITUDE OF ASCENDING NODE
	TA	I	TRUE ANOMALY
	R(3)	O	POSITION VECTOR IN REFERENCE SYSTEM
	RM	O	POSITION MAGNITUDE
	V(3)	O	VELOCITY VECTOR IN REFERENCE SYSTEM
	VM	O	VELOCITY MAGNITUDE
	TFP	O	TIME FROM PERIAPSIS

SUBROUTINES SUPPORTED: DATAS VMP NONLIN COPINS NONINS
DATA HELIO MULTAR CPROP

SUBROUTINES REQUIRED: NONE

LOCAL SYMBOLS:	AUXF	ECCENTRIC ANOMALY (HYPERBOLIC CASE)
	AVA	MEAN ANOMALY (ELLIPTIC CASE)
	CI	COSINE OF INCLINATION
	CK	VELOCITY FACTOR USED TO CALCULATE FINAL VELOCITY VECTOR
	CN	COSINE OF LONGITUDE OF ASCENDING NODE
	COSEA	COSINE OF ECCENTRIC ANOMALY (ELLIPTIC CASE)
	CT	COSINE OF TRUE ANOMALY

CW	COSINE OF SUM OF ARGUMENT OF PERIAPSIS AND TRUE ANOMALY/ COSINE OF ARGUMENT OF PERIAPSIS
DIV	THE SUM 1.+E*(COS(TA/RAD)). USED AS A DIVISOR IN SUBSEQUENT EQUATIONS TO CALCULATE TFP
EA	ECCENTRIC ANOMALY (ELLIPTIC CASE)
P	SEMI-LATUS RECTUM
RAD	DEGREES TO RADIANS CONVERSION FACTOR
SINEA	SINE OF ECCENTRIC ANOMALY (ELLIPTIC CASE)
SINHF	HYPERBOLIC SINE OF AUXF
SI	SINE OF INCLINATION
SN	SINE OF LONGITUDE OF ASCENDING NODE
ST	SINE OF TRUE ANOMALY
SW	SINE OF THE SUM OF ARGUMENT OF PERIAPSIS AND TRUE ANOMALY/ SINE OF ARGUMENT OF PERIAPSIS
TANG	INTERMEDIATE VARIABLE USED TO CALCULATE SINHF

ELCAR Analysis

ELCAR transforms the standard conic elements of a massless point referenced to a gravitational body to cartesian position and velocity components with respect to that body.

Let the gravitational constant of the body be denoted μ and the given conic elements $(a, e, i, \omega, \Omega, f)$. The semilatus rectum p is

$$p = a\,(1 - e^2) \tag{1}$$

Then the magnitude of the radius vector is given by

$$r = \frac{p}{1 + e \cos f} \tag{2}$$

The unit vector in the direction of the position vector is

$$\begin{aligned}
u_x &= \cos(\omega + f)\cos\Omega - \cos i \sin(\omega + f)\sin\Omega \\
u_y &= \cos(\omega + f)\sin\Omega + \cos i \sin(\omega + f)\cos\Omega \\
u_z &= \sin(\omega + f)\sin i
\end{aligned} \tag{3}$$

The position vector $\vec{r}$ is therefore

$$\vec{r} = r\,\hat{u} \tag{4}$$

The velocity vector $\vec{v}$ is given by

$$\begin{aligned}
v_x &= \sqrt{\frac{\mu}{p}}\Big[(e + \cos f)(-\sin\omega\cos\Omega - \cos i \sin\Omega\cos\omega) \\
&\qquad -\sin f\,(\cos\omega\cos\Omega - \cos i\sin\Omega\sin\omega)\Big] \\
v_y &= \sqrt{\frac{\mu}{p}}\Big[(e + \cos f)(-\sin\omega\sin\Omega + \cos i\cos\Omega\cos\omega) \\
&\qquad -\sin f\,(\cos\omega\sin\Omega + \cos i\cos\Omega\sin\omega)\Big] \\
v_z &= \sqrt{\frac{\mu}{p}}\Big[(e + \cos f)\sin i\cos\omega - \sin f\sin i\sin\omega\Big]
\end{aligned} \tag{5}$$

The conic time from periapsis t_p is computed from different formulae depending upon the sign of the semi-major axis. For $a > 0$ (elliptical motion)

$$t_p = \sqrt{\frac{a^3}{\mu}}\,(E - e \sin E)$$

$$\cos E = \frac{e + \cos f}{1 + e \cos f} \qquad \sin E = \frac{\sqrt{1 - e^2}\ \sin f}{1 + e \cos f} \tag{6}$$

For $a < 0$ (hyperbolic motion) the time from periapsis is

$$t_p = \sqrt{\frac{a^3}{\mu}}\ (e \sinh H - H)$$

$$\tanh \frac{H}{2} = \sqrt{\frac{e - 1}{e + 1}}\ \tan \frac{f}{2} \tag{7}$$

SUBROUTINE EPHEM

PURPOSE: TO COMPUTE THE CARTESIAN STATE OF DESIRED BODIES AT SPECIFIED TIMES ACCORDING TO TWO OPTIONS:
(1) ECLIPTIC COORDINATES OF ONE BODY RELATIVE TO ITS REFERENCE BODY (SUN FOR PLANETS, EARTH FOR MOON)
(2) ECLIPTIC COORDINATES OF ALL GRAVITATIONAL BODIES RELATIVE TO THE INERTIAL COORDINATE SYSTEM (EITHER HELIOCENTRIC OR BARYCENTRIC).

CALLING SEQUENCE: CALL EPHEM(IC,D,N)

ARGUMENT: D I JULIAN DATE OF REFERENCE TIME (REFERENCED 1950)

IC I FLAG SET EQUAL TO 1 FOR OPTION 1 AND TO 0 FOR OPTION 2

N I NUMBER OF GRAVITATIONAL BODIES TO BE COMPUTED

SUBROUTINES SUPPORTED: HELIO LAUNCH LUNTAR MULCON MULTAR EXCUTE TRAPAR VMP DATAS PCTM PRINT4 PSIM TRAKS GUISIM GUISS PRNTS4 DATA PRINT3 TRAKM GUIDM GUID

SUBROUTINES REQUIRED: CENTER

LOCAL SYMBOLS:

A	SEMI-MAJOR AXIS OF LUNAR CONIC
DD	ONE TEN-THOUSANDTH TIMES THE INPUT ARGUMENT D FOR COMPUTATIONS IN FN1, FN2
E	ECCENTRICITY OF LUNAR CONIC
ECAM	ECCENTRIC ANOMALY USED TO SOLVE KEPLER EQUATION
ECC	ECCENTRICITY USED TO SOLVE KEPLER EQUATION
EM	MEAN ANOMALY OF LUNAR CONIC
E2	E SQUARED
E3	E CUBED
FCTR	VELOCITY DIVIDED BY RADIUS
FN1	STATEMENT FUNCTION DEFINING A THIRD ORDER POLYNOMINAL. USED IN COMPUTATION OF MEAN ANOMALY OF INNER PLANETS AND OF MOON

FN2	STATEMENT FUNCTION DEFINING A FIRST ORDER POLYNOMINAL. USED IN MEAN ANOMALY COMPUTATIONS OF THE OUTER PLANETS
I	INDEX FOR LOGIC CONTROL
IJKL	INCREMENT COUNTER IN SOLUTION OF KEPLER EQUATION
IN	INDEX, ROW OF F OF LUNAR COORDINATES
IND	INDEX, ROW OF F OF COORDINATES OF THE I-TH PLANET
ITEMP	INTERMEDIATE VARIABLE USED TO NORMALIZE CONIC ANGLES
ITEST	INTERNAL CODE WHICH DETERMINES IF COORDINATES OF EARTH ARE BEING CALCULATED IN ORDER TO COMPUTE THOSE OF MOON
ITEST2	INTERNAL CODE WHICH DETERMINES IF COORDINATES OF EARTH HAVE BEEN COMPUTED PRIOR TO COMPUTING THOSE O THE MOON
K	INDEX USED IN CALCULATION OF MEAN ANOMALY
P	SEMI-LATUS RECTUM
PI2	TWO TIMES THE MATHEMATICAL CONSTANT PI
R	HELIOCENTRIC RADIUS OF PLANET
TRG	ARRAY OF TRIGONOMETRIC FUNCTIONS OF SPECIFIED ANGLES
VEL	VELOCITY OF PLANET
WX	X-COMPONENT OF INTERMEDIATE VECTOR, W
WY	Y-COMPONENT OF INTERMEDIATE VECTOR, W
WZ	Z-COMPONENT OF INTERMEDIATE VECTOR, W

COMMON COMPUTED/USED:	ELMNT	F	T	XP	
COMMON USED:	CN	EMN	IBARY	NBODYI	NO
	ONE	PMASS	ST	TWOPI	TWO
	ZERO				

EPHEM Analysis

EPHEM first determines the current value for the mean anomaly of the pertinent body. The mean anomaly M is computed from

$$M = M_0 + M_1 t + M_2 t^2 + M_2 t^3 \qquad \text{for inner planets}$$

$$M = M_0 + M_1 t \qquad \text{for outer planets}$$

$$M = L_0 + L_1 t + L_2 t^2 + L_3 t^3 - \tilde{\omega}(t) \qquad \text{for the moon}$$

Kepler's equation $M = E - e \sin E$ is then solved iteratively to determine the eccentric anomaly E. The subsequent computations are basic conic manipulations:

$$p = a(1 - e^2)$$

$$r = a(1 - e \cos E)$$

$$v = \sqrt{\mu\left(\frac{2}{r} - \frac{1}{a}\right)}$$

$$\cos f = \frac{p - r}{er} \qquad \sin f = \sqrt{1 - \cos^2 f} \quad \operatorname{sgn}(\sin E)$$

$$\cos \gamma = \frac{\sqrt{\mu p}}{rv} \qquad \sin \gamma = \sqrt{1 - \cos^2} \quad \operatorname{sgn}(\sin E)$$

$$\omega = \tilde{\omega} - \Omega$$

The cartesian position and velocity relative to the reference body are then

$$\overline{r} = r_x \hat{i} + r_y \hat{j} + r_z \hat{k}$$

$$r_x = r \cos(\omega + f) \cos \Omega - r \sin(\omega + f) \sin \Omega \cos i$$

$$r_y = r \cos(\omega + f) \sin \Omega + r \sin(\omega + f) \cos \Omega \cos i$$

$$r_z = r \sin(\omega + f) \sin i$$

$$\vec{v} = \frac{v}{r} \left[(\hat{w} \times \vec{r}) \cos \gamma + \vec{r} \sin \gamma\right]$$

where $\hat{w} = (\sin i \sin \Omega)\, \hat{i} - (\sin i \cos \Omega)\, \hat{j} + (\cos i)\, \hat{k}$

When option 1 is used, the reference body for all the planets is the sun while the reference body for the moon is the earth.

When option 2 is used with heliocentric inertial coordinates, the cartesian state of the earth is added to the cartesian state of the moon to convert the state of the moon to heliocentric coordinates before storing that state in the F-array.

When option 2 is used with barycentric inertial coordinates, subroutine CENTER is called to convert all elements to barycentric coordinates before storing in the F-array.

PROGRAM ERRANN

PURPOSE: TO CONTROL THE COMPUTATIONAL FLOW THROUGH THE BASIC CYCLE (MEASUREMENT PROCESSING) AND ALL EVENTS IN THE ERROR ANALYSIS MODE.

SUBROUTINES SUPPORTED: ERRON

SUBROUTINES REQUIRED:

SCHED	NTM	PSIM	DYNO	TRAKM
MEMO	NAVM	PRINT3	SETEVN	GUIDM

LOCAL SYMBOLS:

AY	DUMMY VARIABLE
ICODE	EVENT CODE
IPRN	MEASUREMENT COUNTER FOR PRINTING
MMCODE	MEASUREMENT CODE
NEVENT	EVENT COUNTER
NR	NUMBER OF ROWS IN THE OBSERVATION MATRIX
TRTM2	TIME OF THE MEASUREMENT

COMMON COMPUTED/USED:

ICODE	MCNTR	RI	TEVN	TRTM1
XF	XI			

COMMON COMPUTED: DELTM

COMMON USED:

FNTM	IEVNT	IPRINT	ISTMC	NEV
NMN	NR	NTMC	RF	TEV

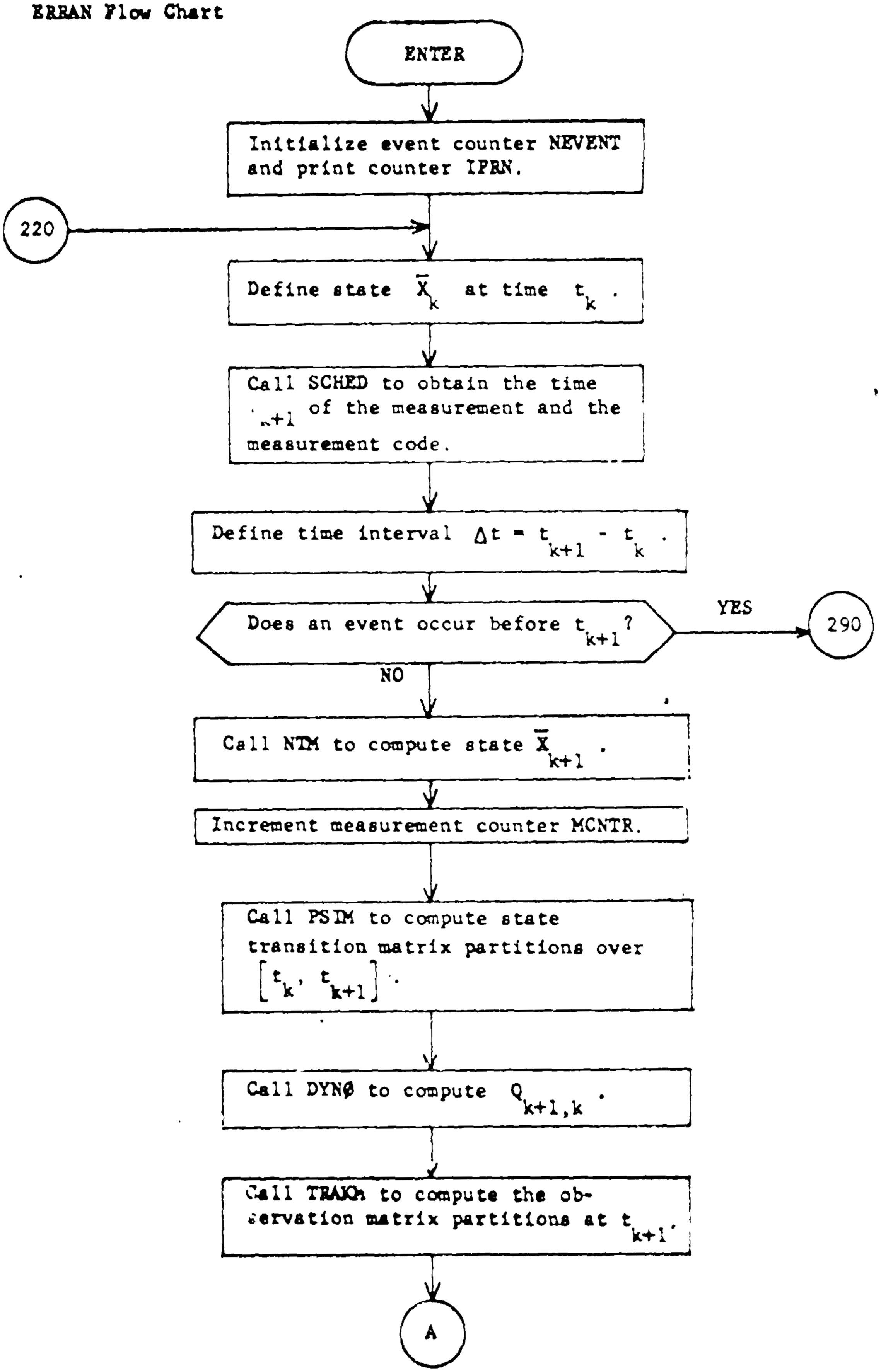
ERRAN Analysis

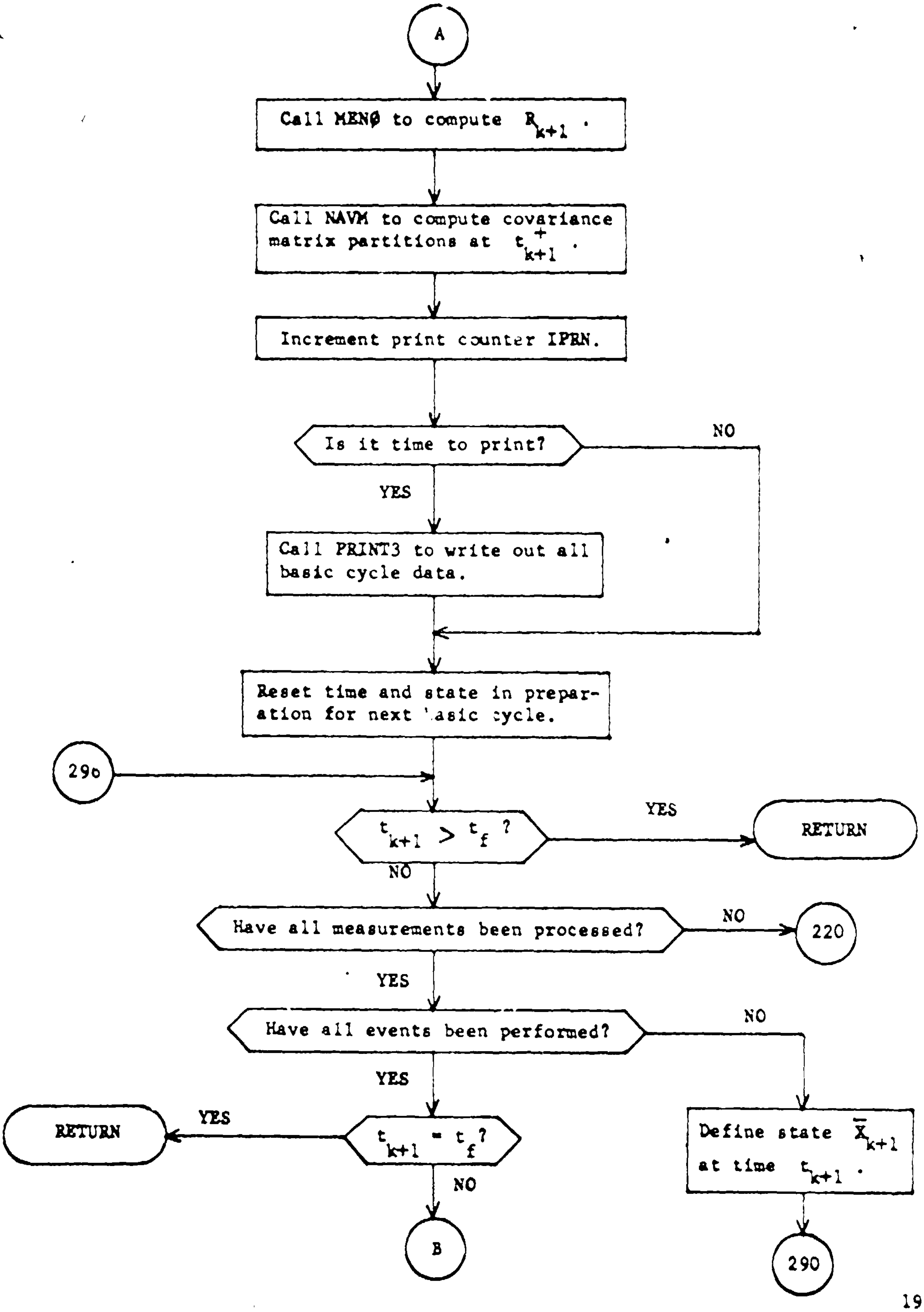
Subroutine ERRAN controls the computational flow through the basic cycle (measurement processing) and all events in the error analysis mode.

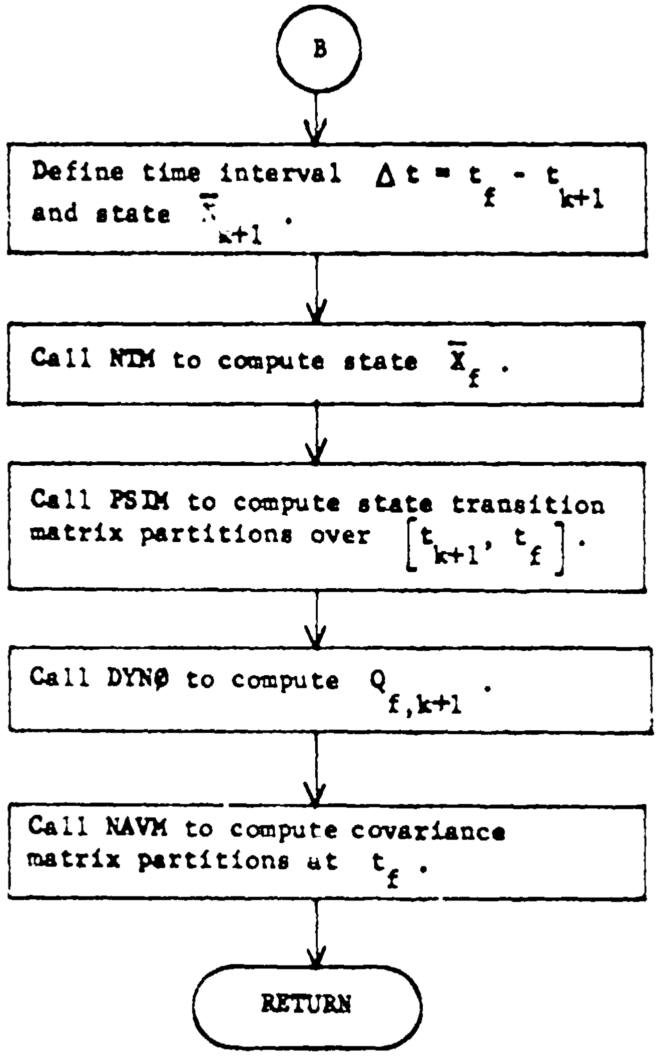
In the basic cycle the first task of ERRAN is to control the generation of the targeted nominal spacecraft state $\overline{X}_{k+1}$ at time t_{k+1}, given the state $\overline{X}_k$ at time t_k. Then, calling PSIM, DYNØ, TRAKM, and MENØ, successively, ERRAN controls the computation of all matrix information required by subroutine NAVM in order to compute the covariance matrix partitions at time t_{k+1}^{+} immediately following the measurement.

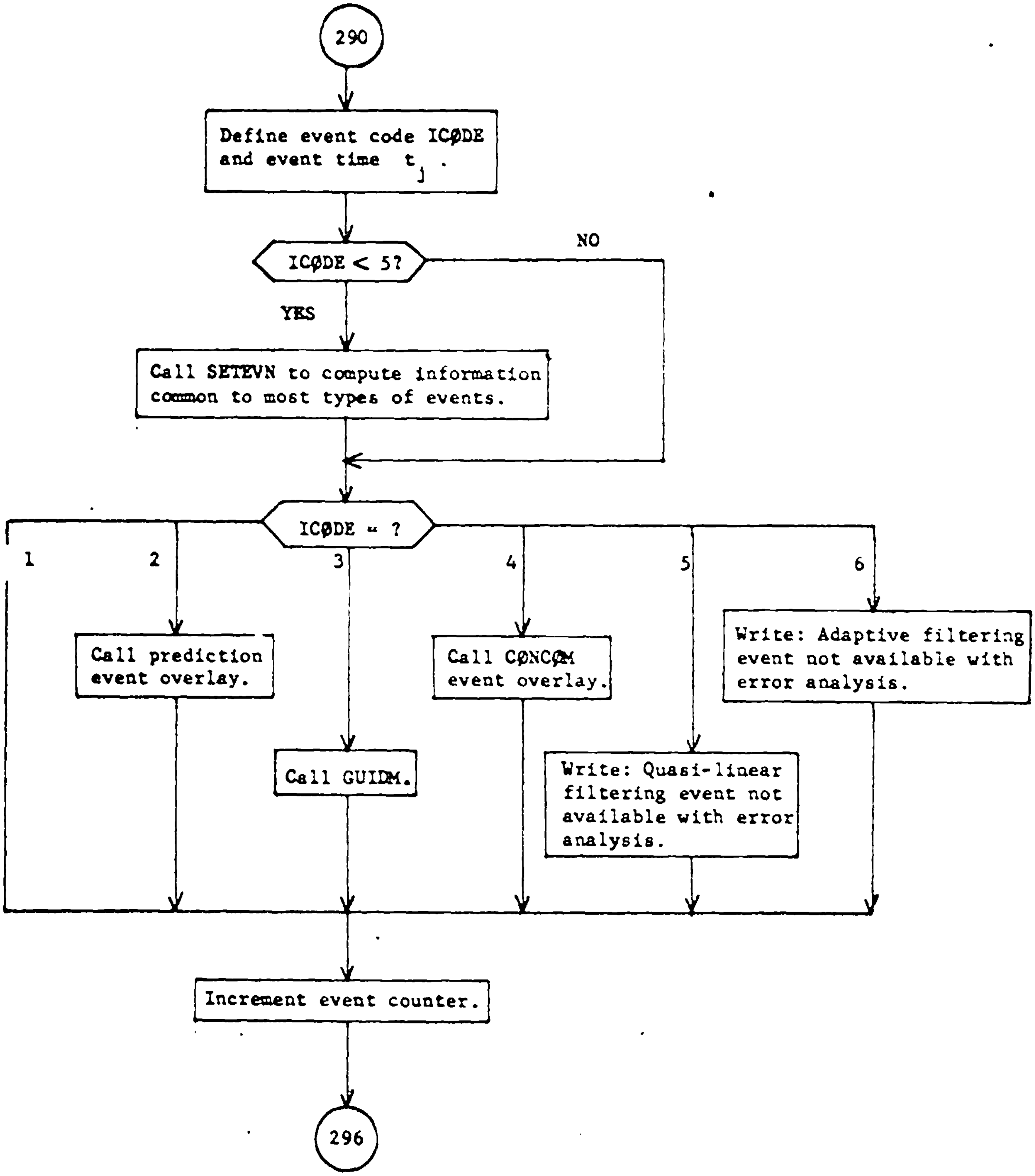
At an event ERRAN simply calls the proper event subroutine or overlay where all required computations are performed.

ERRAN Flow Chart




A
Call MENØ to compute R_{k+1}.
Call NAVM to compute covariance matrix partitions at t^{+}_{k+1}.
Increment print counter IPRN.
Is it time to print?
NO
YES
Call PRINT3 to write out all basic cycle data.
Reset time and state in preparation for next basic cycle.
290
$t_{k+1} > t_f$?
YES
RETURN
NO
Have all measurements been processed?
NO
220
YES
Have all events been performed?
NO
YES
Define state $\bar{X}_{k+1}$ at time t_{k+1}.
290
RETURN
YES
$t_{k+1} = t_f$?
NO
B


B
Define time interval $\Delta t = t_f - t_{k+1}$ and state $\overline{X}_{k+1}$.
Call NTM to compute state $\overline{X}_f$.
Call PSIM to compute state transition matrix partitions over $[t_{k+1}, t_f]$.
Call DYNØ to compute $Q_{f,k+1}$.
Call NAVM to compute covariance matrix partitions at t_f .
RETURN


290
Define event code ICØDE and event time t_j .
ICØDE < 5?
NO
YES
Call SETEVN to compute information common to most types of events.
ICØDE = ?
1
2
3
4
5
6
Call prediction event overlay.
Call GUIDM.
Call CØNCØM event overlay.
Write: Quasi-linear filtering event not available with error analysis.
Write: Adaptive filtering event not available with error analysis.
Increment event counter.
296

PROGRAM ERRON

PURPOSE: TO CONTROL THE ERROR ANALYSIS OVERLAY SCHEME

SUBROUTINES SUPPORTED: NONE

SUBROUTINES REQUIRED: DATA ERRAN PRNTS3

LOCAL SYMBOLS: IRUNX TOTAL NUMBER OF DATA CASES

IRUN DATA CASE COUNTER

SUBROUTINE ESTMT

PURPOSE: TO UPDATE THE FINAL VALUES OF THE PRECEDING COMPUTATION INTERVAL WHICH SERVE AS INITIAL VALUES FOR THE NEW STEP, TO DETERMINE THE DESIRED SIZE OF THE NEXT TIME INCREMENT ON THE BASIS OF TRUE ANOMALY OR REQUESTED PRINTTIME, AND TO ESTIMATE THE FINAL POSITION AND MAGNITUDE OF THE VIRTUAL MASS.

CALLING SEQUENCE: CALL ESTMT(D1,DELTM,TRTM)

ARGUMENTS	D1	I	JULIAN DATE, EPOCH 1900, OF THE INITIAL TRAJECTORY TIME
	DELTM	I	TIME INTERVAL OVER WHICH THE TRAJECTORY WILL BE PROPAGATED (DAYS)
	TRTM	I	INITIAL TRAJECTORY TIME (DAYS) REFERENCED TO INJECTION

SUBROUTINES SUPPORTED: VMP

SUBROUTINES REQUIRED: NONE

COMMON COMPUTED/USED:	INCMNT	V	
COMMON COMPUTED:	ITRAT	KOUNT	
COMMON USED:	INCPR	INC	IPR

ESTMT Analysis

The initial values of the state variables are first set equal to the values at the end of the previous interval. The nominal time interval to be used during the current step is computed from

$$\Delta t_k = \frac{c_2 \, r_{VS_B}}{V_{VS_B}} \tag{1}$$

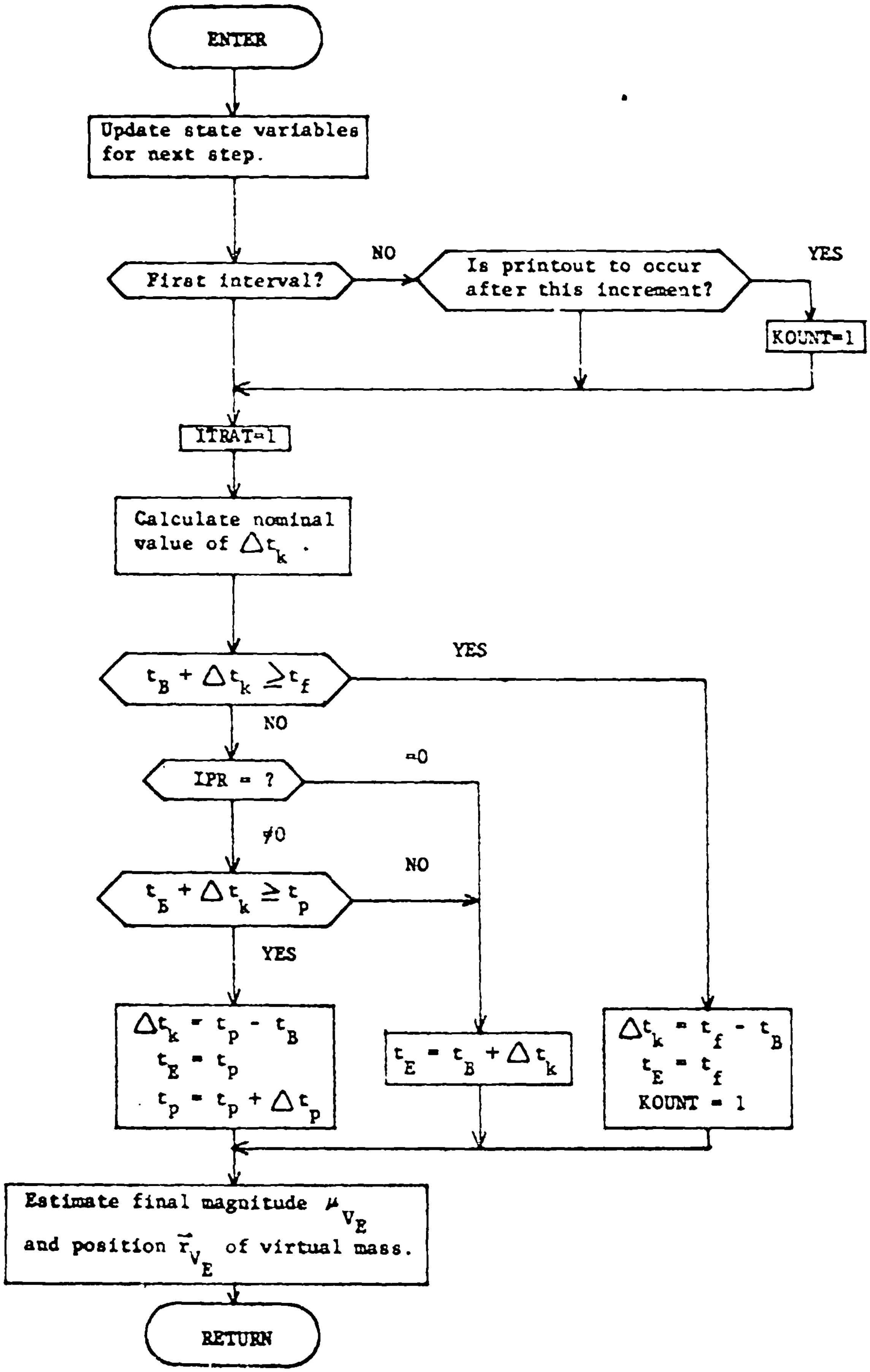
where c_2 is the constant input true anomaly increment relative to the virtual mass trajectory.

The time interval to the final time t_f or to the next time printout t_p is computed and the current time interval Δt is adjusted if necessary.

Finally the virtual mass final position and magnitude are estimated by the expansions

$$\begin{aligned} \mu_{V_E} &= \mu_{V_B} + \dot{\mu}_{V_B} \Delta t + \bar{\bar{\mu}}_V \Delta t^2 \\ \vec{r}_{V_E} &= \vec{r}_{V_B} + \dot{\vec{r}}_{V_B} \Delta t + \bar{\bar{\vec{r}}}_{V_{av}} \Delta t^2 \end{aligned} \tag{2}$$

ESTMT Flow Chart



SUBROUTINE EULMX

PURPOSE: TO COMPUTE THE MATRIX REQUIRED TO DEFINE TRANSFORMATIONS FROM ONE COORDINATE SYSTEM TO ANOTHER.

CALLING SEQUENCE: CALL EULMX(ALP,NN,BET,MM,GAM,LL,P)

ARGUMENT:	ALP	I	FIRST ROTATION ANGLE (RADIANS)
	NN	I	FIRST AXIS OF ROTATION
	BET	I	SECOND ROTATION ANGLE (RADIANS)
	MM	I	SECOND AXIS OF ROTATION
	GAM	I	THIRD ROTATION ANGLE (RADIANS)
	LL	I	THIRD AXIS OF ROTATION
	P(3,3)	O	TRANSFORMATION MATRIX

SUBROUTINES SUPPORTED: PECEQ

SUBROUTINES REQUIRED: NONE

LOCAL SYMBOLS:	A	INTERMEDIATE ROTATION MATRIX
	ALPHA	TEMPORARY LOCATION FOR EACH OF THE ROTATION ANGLES: ALP, BET, AND GAM
	D	INTERMEDIATE PRODUCT MATRIX
	F	TRANSFORMATION MATRIX FOR ANGLE ALP
	G	TRANSFORMATION MATRIX FOR ANGLE BET
	H	TRANSFORMATION MATRIX FOR ANGLE GAM
	N	COUNTER SHOWING NUMBER OF COORDINATE AXES FOR WHICH CALCULATIONS REMAIN
	NAXIS	TEMPORARY LOCATION FOR EACH OF THE AXES OF ROTATION: NN,MM, AND LL

COMMON USED: ONE ZERO

SUBROUTINE EXCUT

PURPOSE CONTROL EXECUTION OF A VELOCITY CORRECTION MODELED AS AN IMPULSE SERIES IN THE ERROR ANALYSIS PROGRAM

CALLING SEQUENCE: CALL EXCUT

SUBROUTINES SUPPORTED: GUIDM

SUBROUTINES REQUIRED: PREPUL PULCOV PULSEX

COMMON COMPUTED/USED: XXIN

COMMON COMPUTED: QK

COMMON USED: DELPX, DIPX, TM, INPX

SUBROUTINE EXCUTE

PURPOSE: TO CONTROL THE ACTUAL EXECUTION OF THE VELOCITY INCREMENT DELTAV.

CALLING SEQUENCE: CALL EXCUTE

SUBROUTINES SUPPORTED: GIDANS

SUBROUTINES REQUIRED: PREPUL PULSEX CAREL PECEQ

LOCAL SYMBOLS:

A	SEMIMAJOR AXIS OF DOMINANT BODY CONIC
DVM	MAGNITUDE OF VELOCITY INCREMENT
E	ECCENTRICITY OF DOMINANT BODY CONIC
INDEX	CODE OF BODY BEING TESTED FOR DOMINANT BODY
IND	INDEX OR CODE OF DOMINANT BODY
ISUN	SUN VALUE OF IND
I	INDEX
JX	INDEX OF S/C-REL-TO-BODY ROW OF F-ARRAY
MODEL	EXECUTION MODEL (1=IMPULSIVE, 2=PULSE ARC)
PP	UNIT VECTOR TO PERIAPSIS IN ORBITAL PLANE
QQ	UNIT VECTOR NORMAL TO PP IN ORBITAL PLANE
RN	POSITION AND VELOCITY OF S/C AT END OF EXECUTION BY PULSING ARC
RSI	POSITION VECTOR OF S/C RELATIVE TO DOMINANT BODY AT EXECUTION TIME
RTB	RADIUS MAGNITUDE TO BODY BEING TESTED FOR DOMINANT BODY
TA	TRUE ANOMALY ON DOMINANT BODY CONIC
TFP	TIME FROM PERIAPSIS ON DOMINANT BODY CONIC
VSI	VELOCITY VECTOR OF S/C RELATIVE TO DOMINANT BODY AT EXECUTION TIME
WW	UNIT NORMAL TO ORBITAL PLANE
W	ARGUMENT OF PERIAPSIS OF DOMINANT BODY

	CONIC	
	XI	INCLINATION OF DOMINANT BODY CONIC
	XMU	GRAVITIONAL CONSTANT OF DOMINANT BODY
	XN	LONGITUDE OF ASCENDING NODE OF DOMINANT BODY

COMMON COMPUTED/USED:	DELTAV	D1	RIN	TRTM	
COMMON COMPUTED:	KTIM				
COMMON USED:	ALNGTH	DELV	F	KUR	MOL
	NBOD	NB	PMASS	PULT	SPHERE
	TM	TWO	V		

EXCUTE Analysis

EXCUTE is the executive subroutine controlling the actual execution of the velocity increment $\overrightarrow{\Delta v}$. The $\overrightarrow{\Delta v}$ is computed by TARGET or INSERS or read in by the user.

Before executing the correction EXCUTE computes peripheral information of interest to the user. It first determines the dominant body acting on the spacecraft. If the spacecraft is in the moon's SOI (with respect to the earth), the moon is the dominant body. If not in the moon's SOI but in any of the planets' SOI (with respect to the sun) that planet is the dominant body. Otherwise the sun is the dominant body.

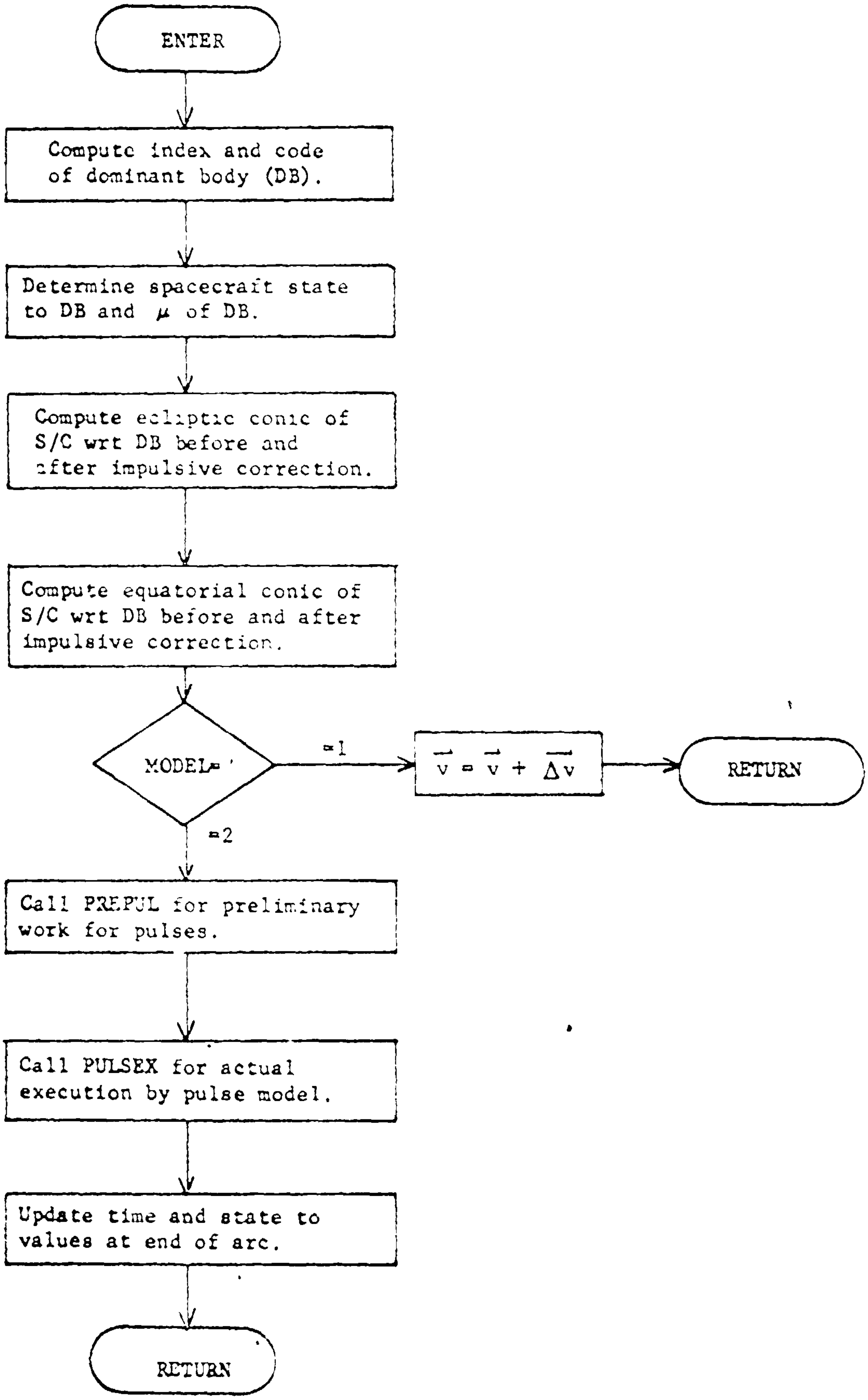
Having determined the dominant body EXCUTE computes the state of the spacecraft relative to that body. It then computes the conic elements of the trajectory both before and after an impulsive addition of the $\overrightarrow{\Delta v}$ in ecliptic coordinates.

If the dominant body is not the sun, it makes the same computations in equatorial coordinates.

EXCUTE then operates on the current value MODEL of the array MDL. If MODEL = 1, the impulsive model of execution is commanded. The $\overrightarrow{\Delta v}$ is therefore added to the current inertial ecliptic velocity before returning to GIDANS.

If MODEL = 2, the pulsing arc model of execution is required. PREPUL is called to perform the preliminary work needed for the pulsing arc. PULSEX then actually propagates the trajectory through the series of pulses. At the completion of the arc EXCUTE updates the time and inertial ecliptic state (both position and velocity) of the nominal trajectory to the state determined by PULSEX.

EXCUTE Flow Chart



SUBROUTINE EXCUTS

PURPOSE CONTROL EXECUTION OF A VELOCITY CORRECTION MODELED AS AN IMPULSE SERIES IN THE SIMULATION PROGRAM

CALLING SEQUENCE: CALL EXCUTS

SUBROUTINES SUPPORTED: GUISIM

SUBROUTINES REQUIRED: PREPUL, PULCOV, PULSEX

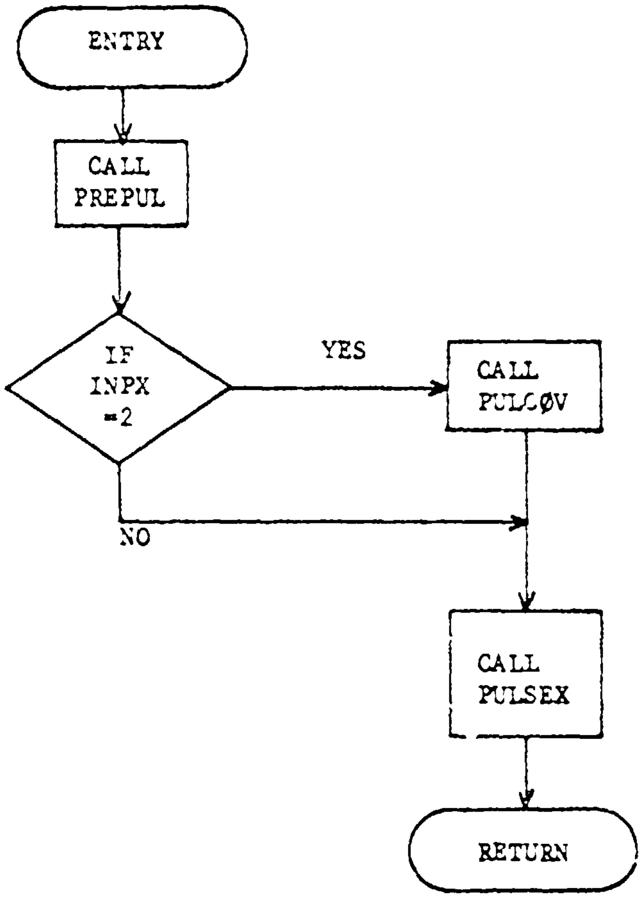
LOCAL SYMBOLS RN EFFECTIVE SPACECRAFT STATE AFTER A VELOCITY CORRECTION MODELED AS AN IMPULSE SERIES

COMMON COMPUTED/USED: XXIN

COMMON COMPUTED: QK

COMMON USED: DELPX, DIPX, TM, INPX

EXCUTS Flow Chart



SUBROUTINE FLITE

PURPOSE: TO SOLVE THE TIME OF FLIGHT EQUATION (LAMBERT-S THEOREM) USING BATTIN-S UNIVERSAL EQUATION FORMULATION.

CALLING SEQUENCE: CALL FLITE(R1,R2,THETA,GM,TF,A,E,K)

ARGUMENTS:

R1	I	INITIAL RADIUS
R2	I	FINAL RADIUS
THETA	I	CENTRAL ANGLE
GM	I	GRAVITIONAL CONSTANT
TF	I	TIME OF FLIGHT
A	O	SEMIMAJOR AXIS
E	O	ECCENTRICITY
K	O	ERROR CODE =0 NO ERROR =1 ERROR

SUBROUTINES SUPPORTED: HELIO

SUBROUTINES REQUIRED: NONE

LOCAL SYMBOLS:

AMISS	ERROR IN ITERATE
BIGNO	CONSTANT = E+25
B1	CONSTANT = S 3/2
CHECK	ERROR IN ITERATE
CX	BATTIN C-FUNCTION OF X
CY	BATTIN C-FUNCTION OF Y
C	CHORD LENGTH
DEM	INTERMEDIATE VARIABLE
P	SEMILATUS RECTUM
ROOT	INTERMEDIATE VARIABLE
SLOP	VALUE OF DERIATIVE OF T(X)
SX	BATTIN S-FUNCTION OF X

SY	BATTIN S-FUNCTION OF Y
S1	SEMIPERIMETER
S	=INTERMEDIATE VARIABLE (=1-C/S1)
TIME	FLIGHT TIME CORRESPONDING TO ITERATE X
T	NORMALIZED TIME OF FLIGHT
U	FLAG SET TO 1 IF X LESS THAN PI 2,-1 ELSE
VB1	INTERMEDIATE VARIABLE
V	FLAG SET TO 1 FOR TYPE I,-1 FOR TYPE II
X1	STARTING VALUE FOR X
X	VARIABLE INTRODUCED TO REPLACE A
Y	INTERMEDIATE VARIABLE AS FUNCTION OF X

FLITE Analysis

FLITE solves the time of flight equation (Lambert's theorem) using Battin's universal equation formulation. Stated functionally Lambert's theorem states that the time of flight t_f is a function

$$t_f = t_f\,(r_1 + r_2,\ c,\ a) \tag{1}$$

solely of the sum $r_1 + r_2$ of the distances of the initial and final points of the trajectory from the central body, the length c of the chord joining these points, and the length of the semimajor axis a of the trajectory. Usually the time of flight is known and it is desired to solve for the semi-major axis. The standard formulation involves different equations for the elliptic, parabolic, and hyperbolic cases, all of which then iterate on a to determine the solution.

In Battin's approach the semimajor axis a is replaced by a new variable x. By further introducing two new transcendental functions $S(x)$ and $C(x)$, the special cases of the flight-time equation are combined into one single, better behaved formula. The functions $S(x)$ and $C(x)$ are defined by

$$\begin{aligned}
S(x) &= \frac{\sqrt{x} - \sin\sqrt{x}}{\sqrt{x}^{\,3}} & C(x) &= \frac{1-\cos\sqrt{x}}{x} & x &> 0\\
&= \frac{\sinh\sqrt{-x} - \sqrt{-x}}{\sqrt{-x}^{\,3}} & &= \frac{\cosh\sqrt{-x} - 1}{-x} & x &< 0\\
&= \frac{1}{6} & &= \frac{1}{2} & x &= 0
\end{aligned} \tag{2}$$

A parameter Q is introduced as

$$Q = \frac{s-c}{s}$$

where

$$c = (r_1^{\,2} + r_2^{\,2} - 2r_1 r_2 \cos\theta)^{\frac{1}{2}}$$

$$s = \frac{1}{2}(r_1 + r_2 + c) \tag{3}$$

The universal flight-time formula is

$$T = \frac{S(x)}{C^{3/2}(x)} \mp Q^{3/2} \frac{S(y)}{C^{3/2}(y)}$$

$$yC(y) = Q\,x\,C(x) \qquad (4)$$

where $T = \sqrt{\frac{\mu}{s^3}}\; t_f$. The choice of the upper or lower sign is made according to whether the transfer angle θ is less or greater than 180° respectively.

The development of equations (4) is too long and complex to be given here. It may be obtained from the first reference listed below. The following steps of that reference are noted:

(1) the two body problem on pp. 15,16
(2) the "vis viva" equation and Kepler's equation on pp. 50,51
(3) Lambert's theorem proved from Kepler's equation on p. 71
(4) the basic flight-time formula and detailed analysis on pp. 72-78
(5) The universal formulation on pp. 80,81.

Instead of using the equations (4) the authors of refence 2 (listed below) determined y as a function of x as

$$y = 4 \arcsin^2 \sqrt{\left|\frac{xsC(x)}{2}\right|} \qquad x \geq 0$$

$$= -4 \ln^2 \left\{ \sqrt{\left|\frac{xsC(x)}{2}\right|} + \sqrt{\left|\frac{xsC(x)}{2}\right| + 1} \right\}^{\frac{1}{2}} \qquad x < 0 \qquad (5)$$

Therefore a single variable iteration is possible. Newton's method is used to solve (4a) given T and Q as

$$x = x - \frac{T(x_n) - T}{T'(x_n)} \qquad (6)$$

where $$T(x) = \frac{S(x)}{C^{3/2}(x)} + Q^{3/2} \frac{S(y)}{C^{3/2}(y)} \qquad (7)$$

$$T'(x) = \frac{1 + k\left[\mp Q^{3/2} - 1.5\sqrt{2-yC(y)}\;\; T(x)\right]}{2x\sqrt{C(x)}} \qquad (8)$$

$$k = \operatorname{sgn}(\pi^2 - x) \sqrt{\frac{2-xC(x)}{2-yC(y)}}\ , \tag{9}$$

As $|2-yC(y)| \longrightarrow 0$, $k \longrightarrow 1$. Therefore if $|2-yC(y)| < 10^{-4}$ k is set to 1. Also $T'(x)$ breaks down as $x \longrightarrow 0$. Therefore the approximation is used:

$$T'(x) = \frac{1 \mp Q^{3/2}}{2\pi^2} \qquad |x| < 10^{-6} \tag{10}$$

The starting value for x is given by $x = x_1 - \Delta x(T,Q)$ where

$$x_1 = 82.1678 + 352.8045\, T - (123954.8504\, T^2 + 43904.0083\, T + 13423.6819)^{\frac{1}{2}}$$

$$\Delta x(T,Q) = \mp \left(\frac{2.36}{T^2 + .15} + \frac{3.\text{[illegible]}}{T + .1} \right) (0.3\, Q^2 + 0.7\, Q) \tag{11}$$

To insure that the routine will not fail for large or small values of T certain restrictions on T are built into the program. The nominal value of T is forced to be no larger than 950,000 and no smaller than 10^{-6}. This forces the corresponding limits for x of $-823.0473 \le x \le 39.14553$.

Finally convergence is achieved when $\left| T(x_n) - T \right| < \frac{T}{100000}$.

Having solved for semimajor axis a, the semilatus rectum p is given by

$$p = \frac{1}{2} \left\{ \frac{r_1 r_2 \sin\theta}{c} \sqrt{\frac{1}{s-c} - \frac{1}{2a}} \pm \operatorname{sgn}(t_m - t) \sqrt{\frac{1}{s} - \frac{1}{2a}} \right\}^2 \tag{12}$$

Then the eccentricity e is given by

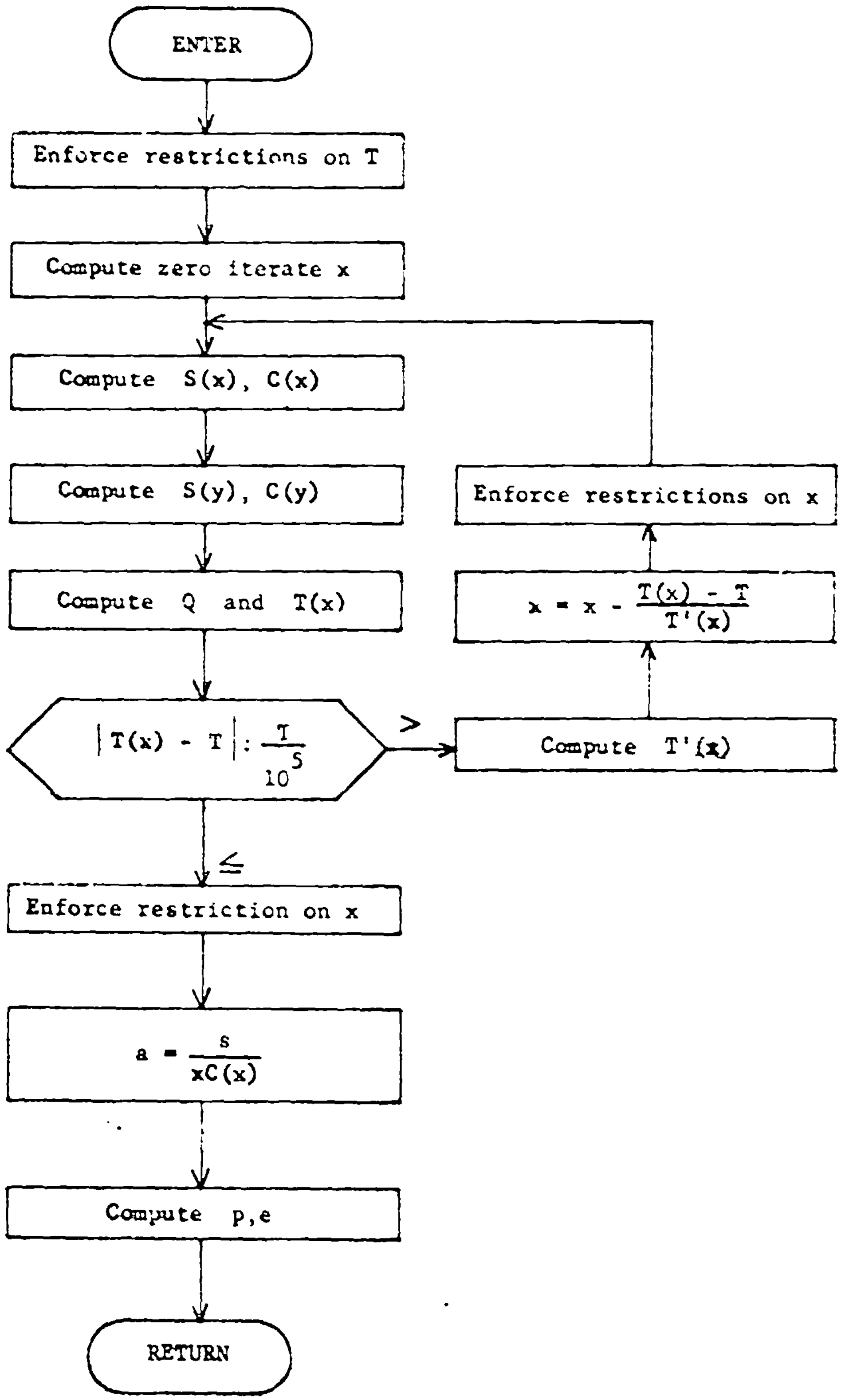
$$e = 1 - \frac{p}{a} \tag{13}$$

References:

(1) Battin, R. H., Astronomical Guidance, McGraw Hill Book Co., Inc., New York, 1964.

(2) Lesh, H. F., and Travis, C., FLIGHT: a Subroutine to Solve the Flight Time Problem, JPL Space Programs Summary 37-53, Vol. II.

FLITE Flow Chart



SUBROUTINE GHA

PURPOSE: TO COMPUTE THE GREENWICH HOUR ANGLE AND THE UNIVERSAL TIME (IN DAYS) WHICH IS USED IN THE TRACKING MODULE TO ORIENT THE TRACKING STATIONS ON A SPERICAL ROTATING EARTH.

CALLING SEQUENCE: CALL GHA

ARGUMENTS: NONE

SUBROUTINES SUPPORTED: DATA1S DATA1

LOCAL SYMBOLS:

D	NUMBER OF DAYS IN TSTAR
EQMEG	EARTH ROTATION RATE
GH	GREENWICH HOUR ANGLE
ID	INTERMEDIATE VARIABLE
REFJD	JULIAN DATE OF JAN. 0, 1950
TFRAC	FRACTION OF DAY IN TSTAR
TSTAR	JULIAN DATE, EPOCH JAN. 0, 1950, OF INITIAL TRAJECTORY TIME

COMMON COMPUTED: UNIVT

COMMON USED: DATEJ EM13

GHA Analysis

Subroutine GHA computes the Greenwich hour angle in degrees and days at some epoch T* referenced to 1950 January 1^d0^h. Epoch T^* is computed from

$$T^* = J.D._0 + 2415020.0 - J.D._{REF}$$

where

$J.D._0$ = Julian date at launch time t_o referenced to 1900 January 0^d12^h.

$J.D._{REF}$ = Reference Julian date 2433282.5

= 1950 January 1^d0^h referenced to January 0^d12^h of the year 4713 B.C.

and 2415020.0 = 1900 January 0^d12^h referenced to January 0^d12^h of the year 4713 B.C.

Then T* is the Julian date at launch time t_o referenced to 1950 January 1^d0^h.

The Greenwich hour angle corresponding to T* is given by

$$GHA(T^*) = 100.0755426 + 0.9856473460d + 2.9015 \times 10^{-13} d^2 + \omega t$$

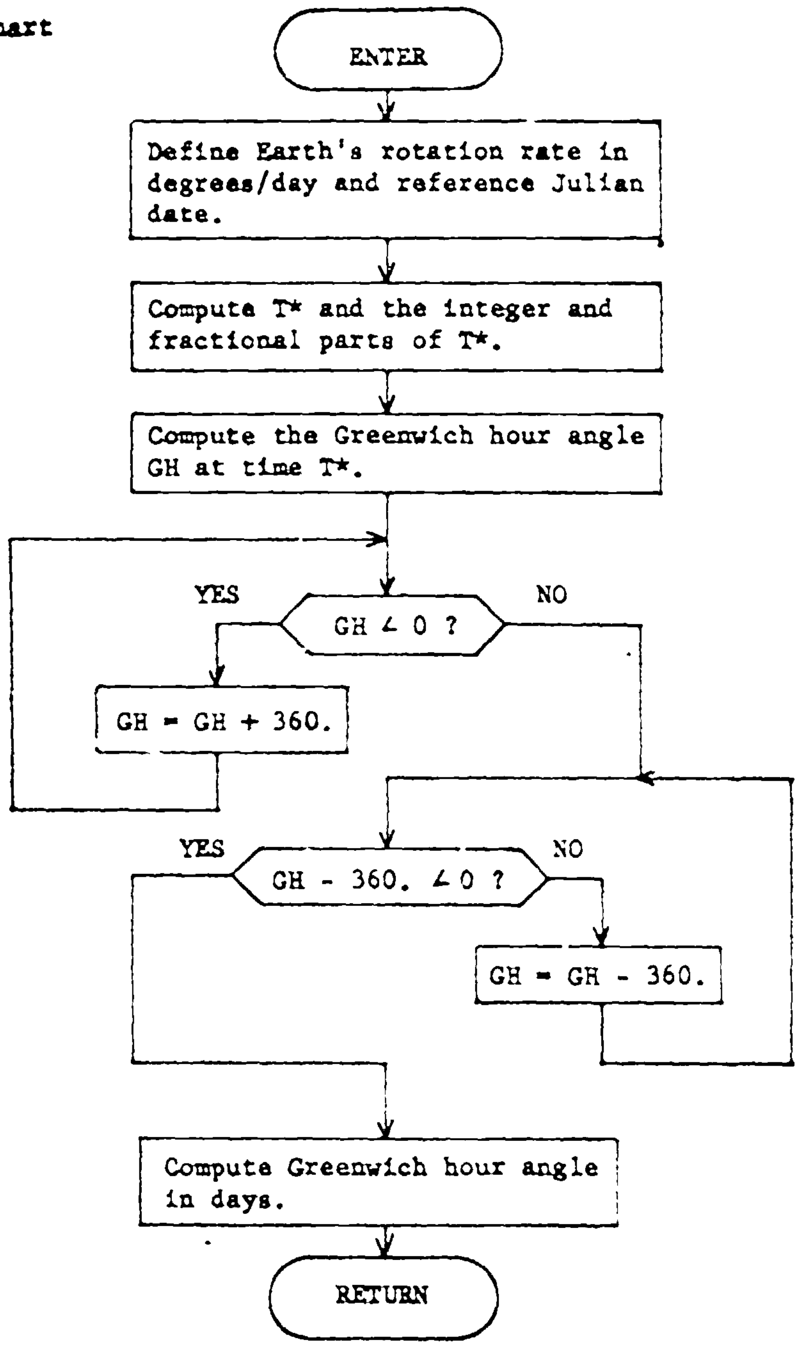
where $0 \leq GHA(T^*) < 360^\circ$.

and d = integer part of T*, t = fractional part of T*,

and ω = Earth's rotation rate is degrees/day.

The Greenwich hour angle in days is given by $\frac{GHA}{\omega}$.

GHA Flow Chart



SUBROUTINE GIDANS

PURPOSE: EXECUTIVE ROUTINE FOR COMPUTATION OF REQUIRED GUIDANCE EVENT

CALLING SEQUENCE: CALL GIDANS

SUBROUTINES SUPPORTED: NOMNAL NONLIN

SUBROUTINES REQUIRED: EXCUTE ZERIT TARGET INSERS TIME VMP

LOCAL SYMBOLS:

	DTIME	DELTA TIME (DAYS) BETWEEN ORBIT INSERTION COMPUTATION AND EXECUTION
	IZER	VECTOR OF CODES USED FOR RETARGETING IZER(KUR)=0, DO NOT RECOMPUTE ZERO ITERATE ≠0, RECOMPUTE ZERO ITERATE
	I	INDEX
	KTYPE	VALUE OF KTYP(KUR) INDICATING TYPE OF EVENT =1, ORIGINAL TARGETING =2, RETARGETING =3, ORBIT INSERTION
	MODEL	DOES NOT APPEAR IN CURRENT VERSION SEE EXECUTE

COMMON COMPUTED/USED: KMXQ KWIT TIMG

COMMON COMPUTED: DELV KTIM ZDAT

COMMON USED: DELTAV KTYP KUR MOL RIN

GIDANS Analysis

GIDANS is an executive routine responsible for processing a guidance maneuver from the computation of the velocity increment $\overrightarrow{\Delta v}$ to the execution of that correction.

Before entry to GIDANS, TRJTRY has computed the index of the current event (KUR) and has integrated the nominal trajectory to the time of the event. GIDANS now evaluates the KUR component of two integer arrays KTYP and KMXQ. The values of these flags determines the operation of GIDANS. The flag KTYP specifies the type of guidance event to be performed while KMXQ prescribes the compute/execute mode to be used according to

KTYP = -1	Termination event	KMXQ = 1	Compute $\overrightarrow{\Delta v}$ only
1	Targeting event	2	Execute $\overrightarrow{\Delta v}$ only
2	Retargeting event	3	Compute and execute $\overrightarrow{\Delta v}$
3	Orbit Insertion	4	Compute but execute $\overrightarrow{\Delta v}$ later

GIDANS first checks for a termination event. If the current index prescribes such an event, the flag KWIT is set to 1 and a return is made to the main program NOMNAL.

In preparation for a normal guidance event, GIDANS calls VMP with the current spacecraft heliocentric state and a time increment of zero to restore the F and V arrays providing the current geometry of spacecraft and planets. If the current event is an execute-only mode, the transfer is made to the execution section of GIDANS for the addition of the pre-set velocity increment.

Otherwise GIDANS interrogates KTYP for the type of maneuver to be computed. For a targeting event, subroutine TARGET is called directly for the computation of the $\overrightarrow{\Delta v}$ necessary to satisfy input target conditions. After calling TARGET the F and V arrays are restored as indicated above.

A retargeting event is defined as a targeting event which requires the computation of a new zero iterate. Thus a retargeting event is an event in which the current nominal state when integrated forward would miss the target conditions badly. Such an event would be the broken-plane correction. For this event TRJTRY stores the current position (and possibly the target position) in the ZDAT array. It then calls ZERIT for the computation of the massless-planets initial velocity consistent with the target conditions. It then operates identically to the targeting event.

The remaining guidance maneuver is the insertion event. GIDANS calls INSERS for the computation of the velocity increment $\overrightarrow{\Delta v}$ and the time interval Δt before it is to be executed.

Subroutines TARGET and INSERS signal trouble to GIDANS via the flag KWIT. If problems are encountered during their operation such as nonconvergence in TARGET or no insertions possible in INSERS KWIT is set to a 1. Otherwise KWIT = 0. Upon return to NOMNAL, if KWIT = 1 the current case will be terminated while KWIT = 0 will allow the current case to continue.

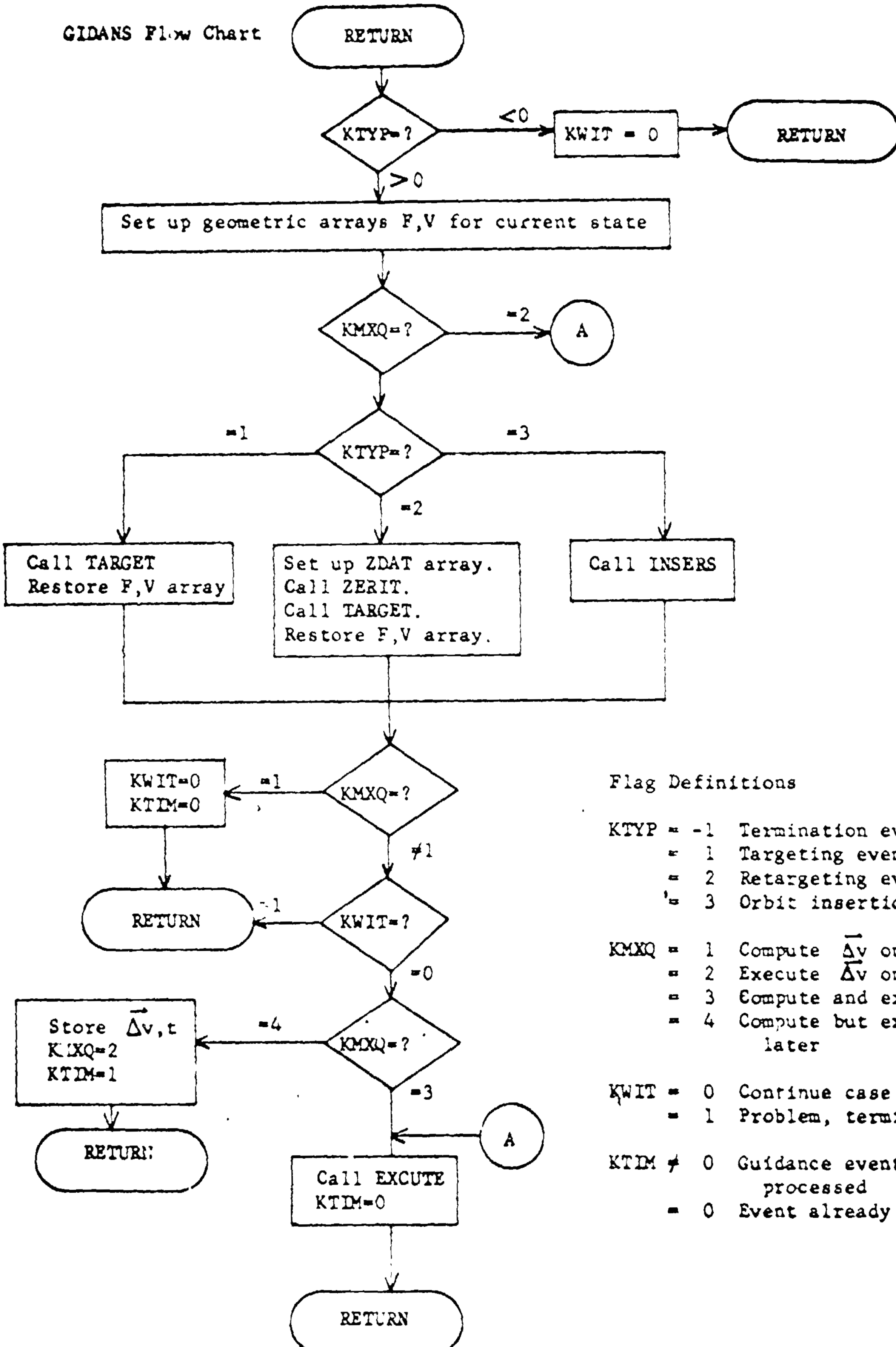
If the current event is a compute-only mode, TRJTRY now sets KWIT = 0 (so that the program will continue regardless of whether the correction computations were successful or not) and returns to NOMNAL. However if the current event failed (KWIT = 1) and was to be executed (KMXQ ≠ 1) GIDANS considers this a fatal error for the current case and returns with KWIT = 1.

If the compute/execute mode is compute-execute later (KMXQ = 4) as is the insertion event, GIDANS now sets up for the subsequent execute-only event. The $\Delta\vec{v}$ computed is stored in the DELV array, the time of the execution is computed ($t_{ex} = t_k + \Delta t$) and stored in the TIMG array, and the KMXQ flag is set to a 2 (execute-only). The return is then made to NOMNAL.

For an event to be executed at the current time (KMXQ = 2,3) GIDANS now calls EXCUTE for the completion of that task.

It should be noted that for all events that are completed at this time the KUR component of the KTIM array are set equal to 0 so that they are no longer considered in determining the next event in TRJTRY. Only in the case of KMXQ = 4 is the KTIM flag non-zero upon exit from GIDANS.

GIDANS Flow Chart



Flag Definitions

KTYP = -1 Termination event
= 1 Targeting event
= 2 Retargeting event
= 3 Orbit insertion

KMXQ = 1 Compute $\vec{\Delta v}$ only
= 2 Execute $\vec{\Delta v}$ only
= 3 Compute and execute $\vec{\Delta v}$
= 4 Compute but execute $\vec{\Delta v}$ later

KWIT = 0 Continue case
= 1 Problem, terminate case

KTIM ≠ 0 Guidance event to be processed
= 0 Event already processed.

SUBROUTINE GUID

PURPOSE COMPUTE GUIDANCE MATRIX, VARIATION MATRIX, AND TARGET CONDITION COVARIANCE MATRIX AT A MIDCOURSE GUIDANCE EVENT IN THE ERROR ANALYSIS PROGRAM

CALLING SEQUENCE: CALL GUID

SUBROUTINES SUPPORTED: GUIDM

SUBROUTINES REQUIRED: EPHEM HYELS JACOBI MATIN NTM ORB PARTL PSIM STMPR VARADA

LOCAL SYMBOLS

A	TWO-VARIABLE B-PLANE GUIDANCE SUB-MATRIX A
BB	TWO-VARIABLE B-PLANE GUIDANCE SUB-MATRIX B
BDRS	B DOT R
BDR1	VALUE OF B DOT R RETURNED FROM PARTL (NOT USED)
BDTS	B DOT T
BDT1	VALUE OF B DOT T RETURNED FROM PARTL (NOT USED)
BS	MAGNITUDE OF B VECTOR
B1	VALUE OF B RETURNED FROM PARTL (NOT USED)
D	INTERMEDIATE JULIAN DATE
DUM1	ARRAY OF EIGENVECTORS
EGVCT	ARRAY OF EIGENVECTORS
EGVL	ARRAY OF EIGENVALUES
ICS	INTERMEDIATE STORAGE FOR ICL2
ICLS	INTERMEDIATE STORAGE FOR ICL
INCMTS	INTERMEDIATE STORAGE FOR INCMT
IPR	INTERMEDIATE STORAGE FOR IPRINT
ISP	INTERMEDIATE STORAGE FOR ISP2
PBR	PARTIAL OF B DOT R WITH RESPECT TO STATE VECTOR
PBT	PARTIAL OF B DOT T WITH RESPECT TO STATE

	VECTOR
PHI1	INTERMEDIATE ARRAY
PHI2	INTERMEDIATE ARRAY
PHI3	INTERMEDIATE ARRAY
RI	NOMINAL SPACECRAFT STATE AT GUIDANCE EVENT
ROW	TARGET CONDITION CORRELATION MATRIX
RTPS	INERTIAL SPACECRAFT STATE AT SPHERE OF INFLUENCE
SQP	TARGET CONDITION STANDARD DEVIATIONS
TCA	TRAJECTORY TIME AT CLOSEST APPROACH
TSI	TRAJECTORY TIME AT SPHERE OF INFLUENCE
XCA	INERTIAL SPECECRAFT STATE AT CLOSEST APPROACH
XSIP	SPACECRAFT [illegible]ON RELATIVE TO TARGET PLANET AT SPHERE OF INFLUENCE
XSIV	SPACECRAFT VELOCITY RELATIVE TO TARGET PLANET AT SPHERE OF INFLUENCE

COMMON COMPUTED/USED:	ICL2	IPRINT	ISPH	ISP2	NO
	XP				
COMMON COMPUTED:	DELTM	EM	TRTM1	TSOI1	
COMMON USED:	ALNGTH	BOR	BDT	B	DATEJ
	DC	DSI	FNTM	FOM	F
	IBARY	ICL	IHYP1	ISTMC	NBOD
	NB	NTMC	NTP	ONE	PHI
	P	RC	RSI	TH	VSI
	ZERO				

GUID Analysis

Subroutine GUID is used in the error analysis mode to compute the same quantities which subroutine GUIS computes in the simulation mode. Subroutine GUID differs from GUIS in that instead of calling NTMS and ABSIM as does GUIS, subroutine GUID calls NTM and VARADA. In addition, the state transition and variation matrices computed in GUID are referenced to the targeted nominal since the most recent nominal is not defined for the error analysis mode. These differences entail only minor logic differences in the flow chart for GUID, and for this reason no GUID flow chart is presented. See subroutine GUIS analysis and flow chart for further details.

SUBROUTINE GUIDM

PURPOSE CONTROL EXECUTION OF A GUIDANCE EVENT IN THE ERROR ANALYSIS PROGRAM

CALLING SEQUENCE: CALL GUIDM

SUBROUTINES SUPPORTED: ERRANN

SUBROUTINES REQUIRED: CORREL DYNO GUID HYELS JACOBI NAVM PSIM STMPR

LOCAL SYMBOLS:

ADA	VARIATION MATRIX
AMAX	INTERMEDIATE VARIABLE USED TO FIND MAXIMUM EIGENVALUE OF VELOCITY CORRECTION COVARIANCE MATRIX (S MATRIX)
CXSU1	STORAGE FOR CXSU KNOWLEDGE COVARIANCE
CXSV1	STORAGE FOR CXSV KNOWLEDGE COVARIANCE
CXU1	STORAGE FOR CXU KNOWLEDGE COVARIANCE
CXV1	STORAGE FOR CXV KNOWLEDGE COVARIANCE
CXXS1	STORAGE FOR CXXS KNOWLEDGE COVARIANCE
DUM1	INTERMEDIATE VARIABLE
DUM	VECTOR SUM OF UPDATE AND STATISTICAL VELOCITY CORRECTIONS
EGM	MAXIMUM EIGENVALUE OF S MATRIX
EGVCT	ARRAY OF EIGENVECTORS
EGVL	ARRAY OF EIGENVALUES
EXEC	EXECUTION ERROR COVARIANCE MATRIX
EXV	EXPECTED VALUE OF VELOCITY CORRECTION
GA	GUIDANCE MATRIX
GAP	INTERMEDIATE ARRAY EQUAL TO GA TIMES P
ICODE	INTERNAL CONTROL FLAG
ICODE2	INTERNAL CONTROL FLAG
IGP	MIDCOURSE GUIDANCE POLICY CODE

IQP	EXECUTION ERROR CODE
ISPHC	TEMPORARY STORAGE FOR ISPH
MAP	INDEX OF MAXIMUM EIGENVALUE OF S
OUT	SPACECRAFT VELOCITY RELATIVE TO TARGET PLANET IN PLANETO-CENTRIC EQUATORIAL COORDINATES
PS1	STORAGE FOR PS KNOWLEDGE COVARIANCE
P1	STORAGE FOR P KNOWLEDGE COVARIANCE
RF	NOMINAL TRAJECTORY STATE AT GUIDANCE EVENT
RHO	MAGNITUDE OF STATISTICAL DELTA-V
ROW	INTERMEDIATE VECTOR
SDV	STANDARD DEVIATION OF MAGNITUDE OF STATISTICAL DELTA-V
SQP	INTERMEDIATE VECTOR
TRS	TRACE OF S MATRIX
U	INTERMEDIATE VARIABLE
VEIG	MATRIX TO BE DIAGONALIZED
Z	INTERMEDIATE ARRAY

COMMON COMPUTED/USED:	CXSUG	CXSU	CXSVG	CXSV	CXUG
	CXU	CXVG	CXV	CXXSG	CXXS
	ISPH	NGE	PG	PSG	PS
	P	TG	XG		
COMMON COMPUTED:	DELTM	TRTM1	XI		
COMMON USED:	FOP	FOV	ICDQ3	ICDT3	IEIG
	IHYP1	ISTMC	NDIM1	NDIM2	NDIM3
	ONE	Q	SIGALP	SIGBET	SIGPRO
	SIGRES	TWO	U0	V0	XF
	ZERO				

GUIDM Analysis

Subroutine GUIDM is the executive guidance subroutine in the error analysis program. In addition to controlling the computational flow for all types of guidance events, GUIDM also performs many of the required guidance computations itself.

Before considering each type of guidance event, the treatment of a general guidance event will be discussed. Let t_j be the time at which the guidance event occurs. Before any guidance event can be executed the targeted nominal state $\bar{X}_j^-$, knowledge covariance $P_{K_j}^-$, and control covariance $P_{c_j}^-$ must all be available, where $(\ )^-$ indicates values immediately before the event. The first two quantities are available prior to entering GUIDM. However, GUIDM controls the propagation of the control covariance over the interval $[t_{j-1}, t_j]$, where t_j denotes the time of the previous guidance event.

The next step in the treatment of a general guidance event is concerned with the computation of the commanded velocity correction and the execution error covariance. In the error analysis program a non-statistical velocity correction is computed whenever the nominal target conditions are changed; otherwise, only a statistical velocity correction can be computed. The commanded velocity correction $\Delta\hat{V}_j$ is then used to compute the execution error covariance matrix $\tilde{Q}_j$. A summary of the execution error model and the equations used to compute $\tilde{Q}_j$ can be found in the subroutine QCØMP analysis section.

The last step is concerned with the updating of required quantities prior to returning to the basic cycle. An assumption underlying the modeled guidance process is that the targeted nominal is always updated by the commanded velocity correction. In the error analysis program only the non-statistical component is used to perform the state update and is indicated by the variable $\Delta\hat{V}_{UP_j}$. Thus, the targeted nominal state immediately following the guidance event is given by

$$\bar{X}_j^+ = \bar{X}_j^- + \begin{bmatrix} 0 \\ \hline \Delta\hat{V}_{UP_j} \end{bmatrix} .$$

The knowledge covariance is updated using the equation

$$P_{K_j}^{+} = P_{K_j}^{-} + \left[\begin{array}{c|c} 0 & 0 \\ \hline 0 & \tilde{Q}_j \end{array}\right]$$

if an impulsive thrust model is assumed. If the thrust is modeled as a series of impulses, then an effective execution error covariance $\tilde{Q}_{eff}$ is computed and the knowledge covariance is updated using the equation

$$P_{K_j}^{+} = P_{K_j}^{-} + \tilde{Q}_{eff} .$$

In either case the control covariance is updated simply by setting

$$P_{c_j}^{+} = P_{K_j}^{+} .$$

This equation is a direct consequence of the assumption that the targeted nominal state is always updated at a guidance event.

A "compute only" option is available in GUIDM in which all of the $( )^{+}$ quantities will still be computed and printed. However, the state and all covariances are then reset to their former $( )^{-}$ values prior to returning to the basic cycle.

Each specific type of guidance event involves the computation of other quantities not discussed above. These will be covered in the following discussion of specific guidance events.

1. Midcourse and Biased Aimpoint Guidance

Linear midcourse guidance policies have form

$$\Delta\hat{V}_{N_j} = \Gamma_j \, \delta\hat{X}_j$$

where the subscript N indicates that this is the velocity correction required to null out deviations from the nominal target state. This notation is required to differentiate between this type of velocity correction and velocity corrections required to achieve an altered target

state. Linear midcourse guidance policies are discussed in more detail in the subroutine GUIS analysis section.

Subroutine GUIDM calls GUID to compute the guidance matrix, Γ_j, and the target condition covariance immediately prior to the guidance event, W_j^-, and then uses Γ_j to compute the velocity correction covariance S_j, which is defined as

$$S_j = E\left[\Delta\hat{V}_{N_j} \ \Delta\hat{V}_{N_j}^{\ T}\right],$$

and is given by the equation

$$S_j = \Gamma_j \ (P_{c_j}^- - P_{K_j}^-)\ \Gamma_j^T .$$

This equation assumes that an optimal estimation algorithm is employed in the navigation process, since the derivation of this equation requires the orthogonality of the estimate and the estimation error.

In the error analysis program $\Delta\hat{V}_{N_j}$ is never available since no estimates $\delta\hat{X}_j$ are ever generated. Only the ensemble statistics of $\delta\hat{X}_j$ are available which means only a statistical or effective velocity correction "$E[\Delta\hat{V}_{N_j}]$" can be computed. In the STEAP error analysis program this effective velocity correction is assumed to have form

$$"E[\Delta\hat{V}_{N_j}]" = \rho_j \frac{\alpha_j}{|\alpha_j|} .$$

The magnitude ρ_j is given by the Hoffman-Young approximation

$$\rho_j = \sqrt{\frac{2A}{\pi}}\left(1 + \frac{B(\pi-2)}{A^2\sqrt{5.4}}\right)$$

where

$$A = \text{trace } S_j = \lambda_1 + \lambda_2 + \lambda_3,$$

$$B = \lambda_1\lambda_2 + \lambda_1\lambda_3 + \lambda_2\lambda_3,$$

and λ_1, λ_2, λ_3 are the eigenvalues of S_j. The direction of the effective velocity correction is assumed to coincide with the eigenvector corresponding to the maximum eigenvalue of S_j. This eigenvector is denoted by α_j. An alternate model assumes the direction coincides with the vector $(\lambda_1, \lambda_2, \lambda_3)$.

If planetary quarantive constraints must be satisfied at a midcourse correction, GUIDM calls BIAIM to compute the new aimpoint μ_j and the (non-statistical) bias velocity correction $\Delta\hat{V}_{B_j}$. All computations in BIAIM are based on linear guidance theory. However, an option is available in GUIDM to recompute $\Delta\hat{V}_{B_j}$, but not μ_j, using nonlinear techniques. This option is recommended if a biased aimpoint guidance event occurs at t_j = injection time. It should also be noted that $\tilde{Q}_j$ is set to zero if t_j = injection time since it is assumed that the injection covariance does not change for small changes in injection velocity.

After the updated control covariance $P^+_{c_j}$ has been computed, the target condition covariance matrix W^+_j following the guidance correction is computed using the equation

$$W^+_j = \eta_j P^+_{c_j} \eta^T_j$$

where variation matrix η_j has been previously computed in subroutine GUID.

2. Re-targeting

In the error analysis (and simulation) program a re-targeting event is defined to be the computation of a velocity correction $\Delta\hat{V}_{RT}$ required to achieve a new set of target conditions using nonlinear techniques. Since the original targeted nominal will be used as the zero-th iterate in the re-targeting process, the new target conditions must be close enough to the original nominal target condition to ensure a covergent process.

It should be noted that after a re-targeting event the new target conditions are henceforth treated as the nominal target conditions.

3. Orbital insertion

An orbital insertion event is divided into a decision event and an execution event. At a decision event the orbital insertion velocity correction $\Delta\hat{V}_{\emptyset I}$ and the time interval Δt separating decision and execution are computed based on the targeted nominal state at t_j. The relevant equations can be found in the subroutine CØPINS analysis section for coplanar orbital insertion; in NØPINS, for non-planar orbital insertion. Before returning to the basic cycle, GUIDM schedules the orbital insertion execution event to occur at $t_j + \Delta t$ and re-orders the necessary event arrays accordingly.

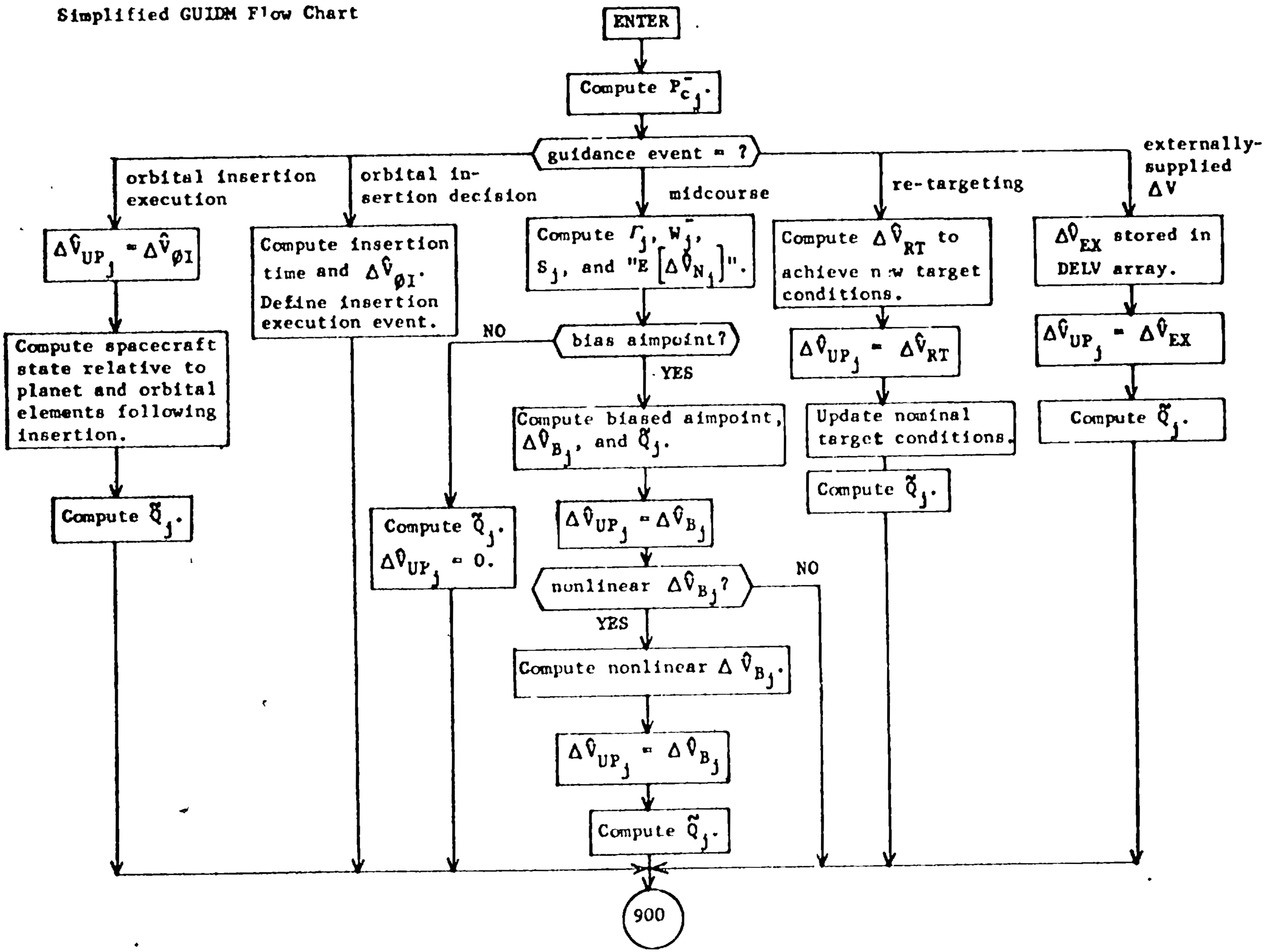
At an orbital insertion execution event the targeted nominal state is updated using the previously computed $\Delta\hat{V}_{\emptyset I}$. In addition, the planeto-centric equatorial components of $\Delta\hat{V}_{\emptyset I}$ and the nominal spacecraft cartesian and orbital element state following the insertion maneuver are computed.

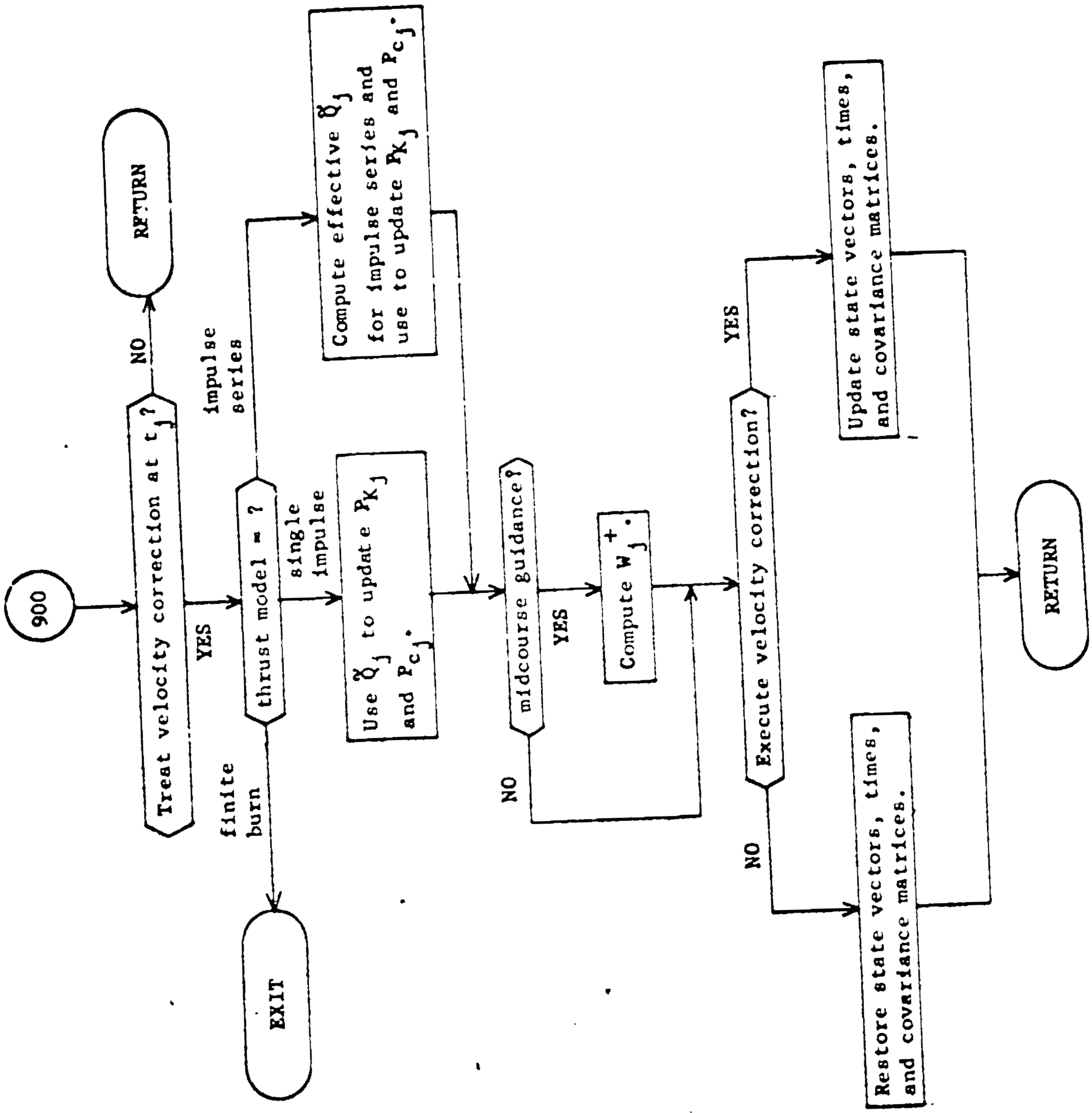
4. Externally-supplied velocity correction

At this type of guidance event the targeted nominal state is simply updated using the externally-supplied velocity correction $\Delta\hat{V}_{EX}$.

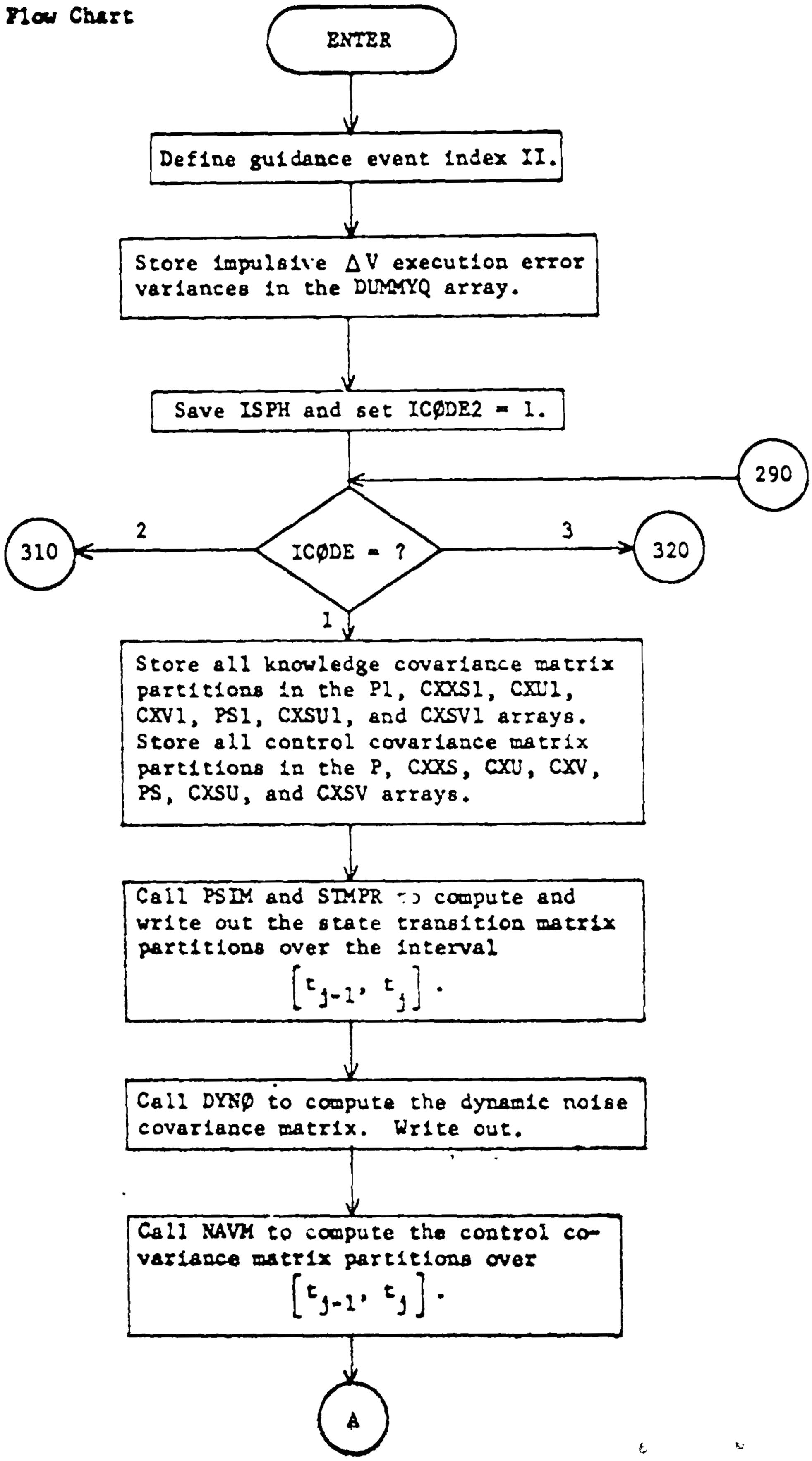
Because of the complexity of the GUIDM flow chart, a simplified flow chart depicting the main elements of the GUIDM structure precedes the complete GUIDM flow chart.

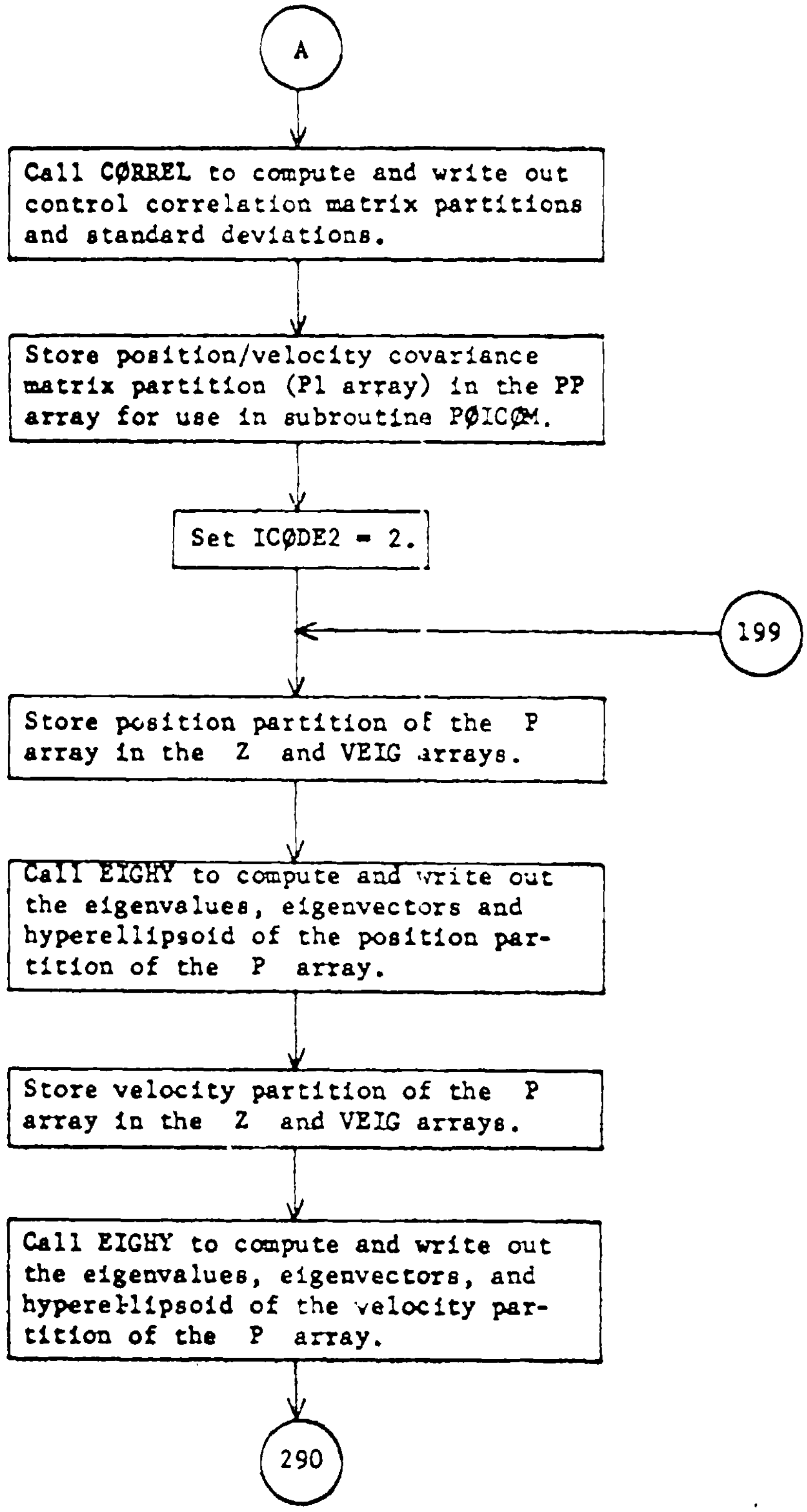
Simplified GUIDM Flow Chart

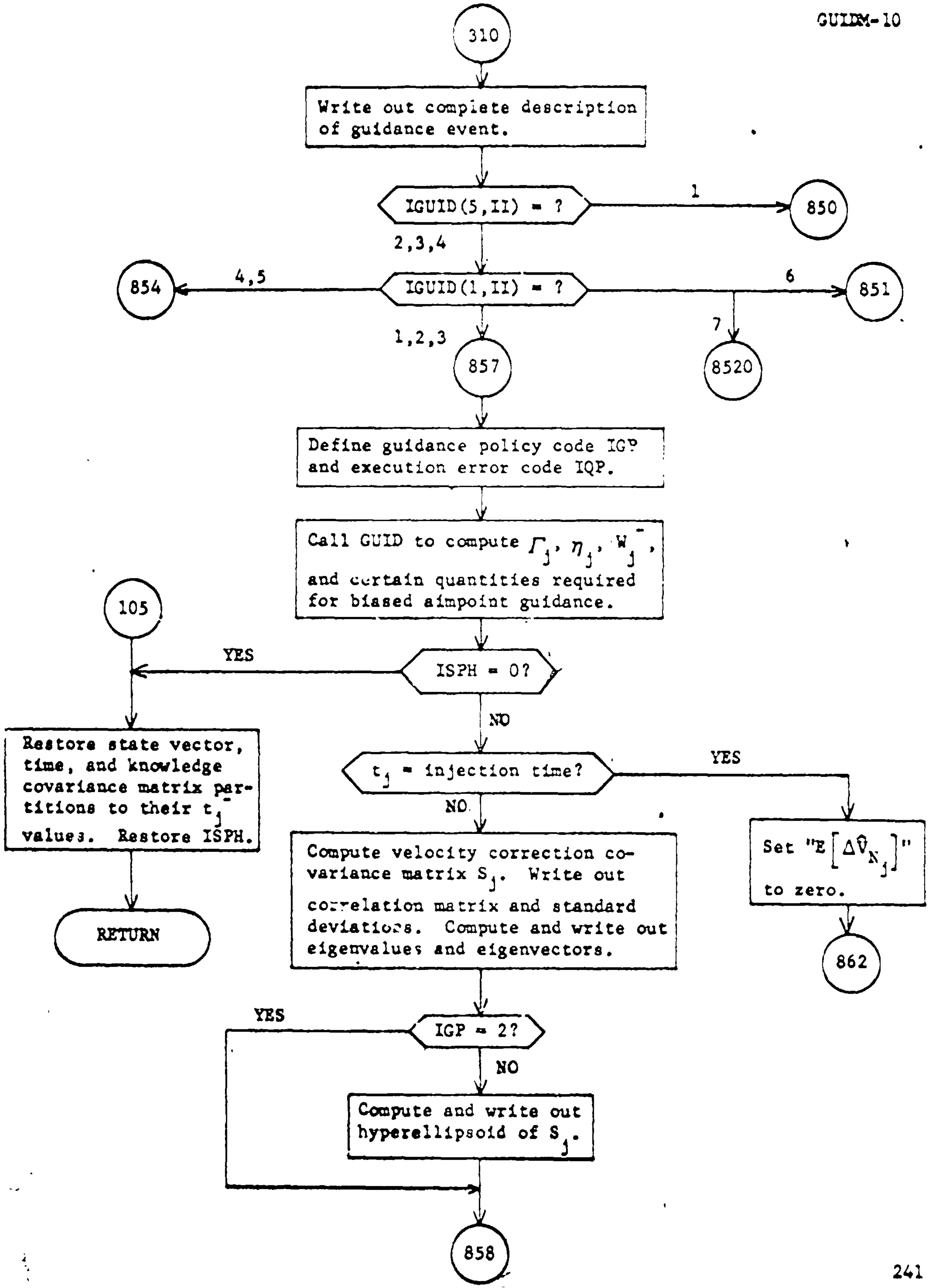


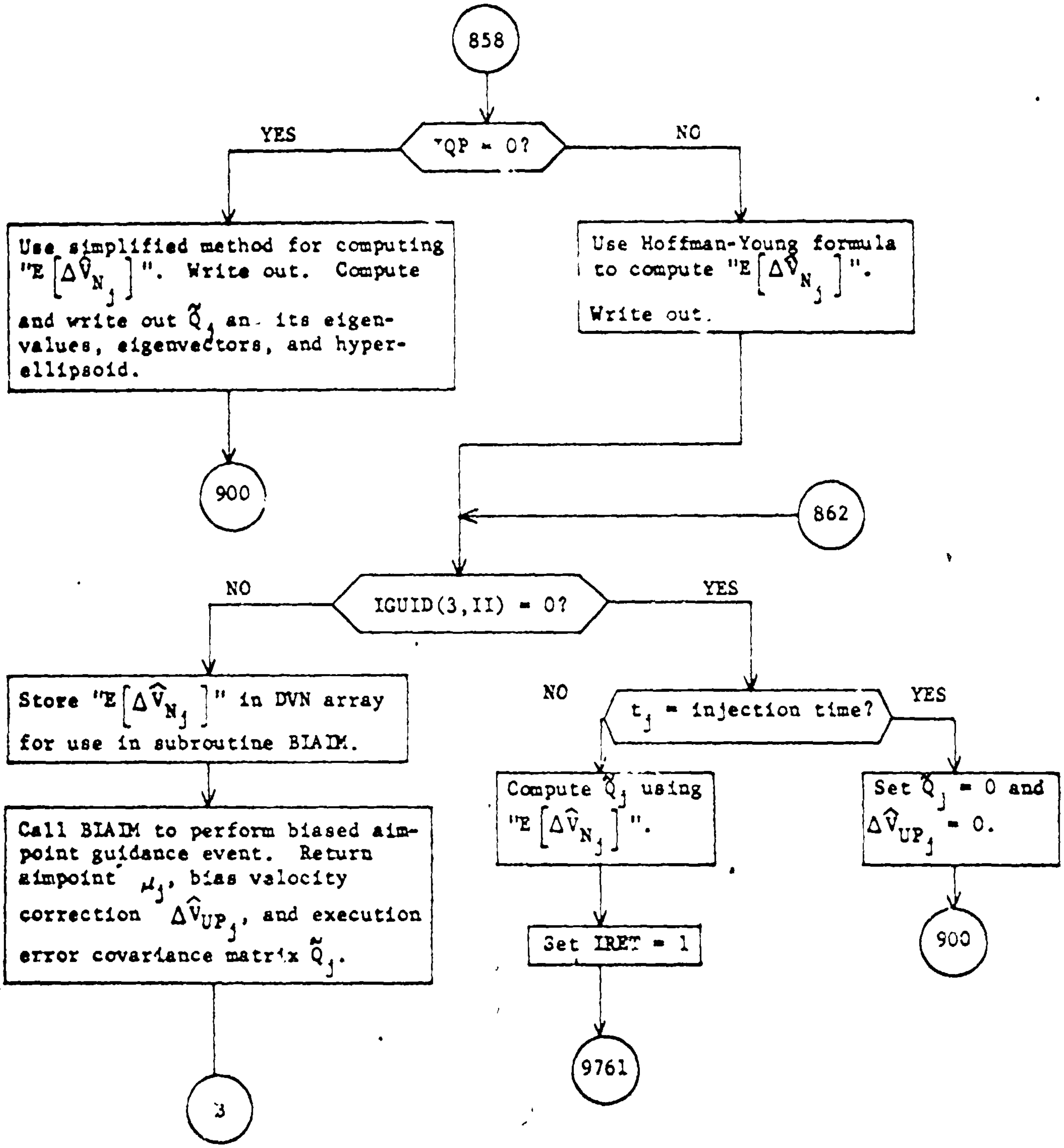

900
Treat velocity correction at t_j?
NO
RETURN
YES
thrust model = ?
impulse series
Compute effective γ_j for impulse series and use to update P_{K_j} and P_{c_j}.
single impulse
Use γ_j to update P_{K_j} and P_{c_j}.
finite burn
EXIT
midcourse guidance?
YES
Compute W_j^+.
NO
Execute velocity correction?
YES
Update state vectors, times, and covariance matrices.
NO
Restore state vectors, times, and covariance matrices.
RETURN

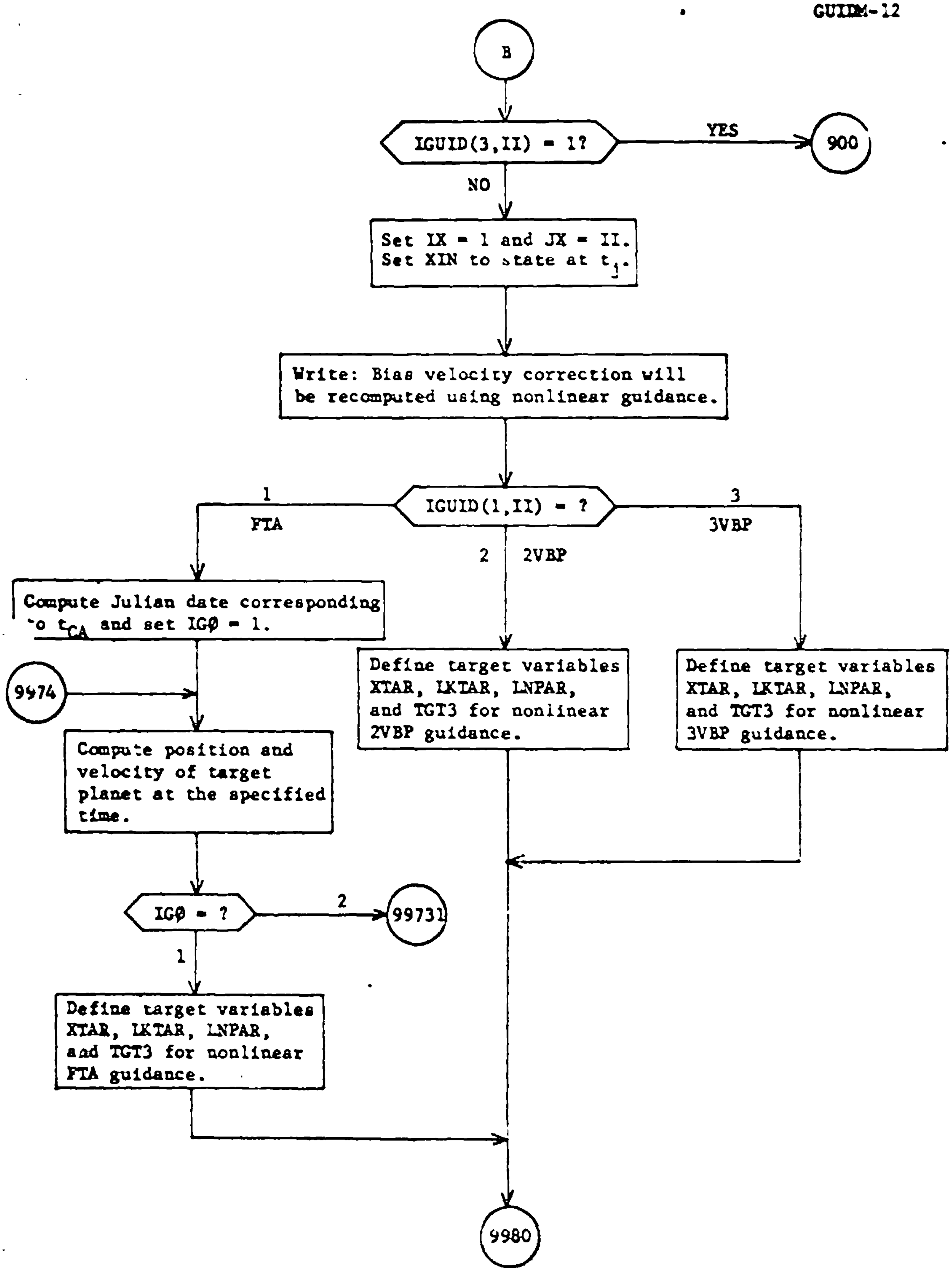
GUIDM Flow Chart

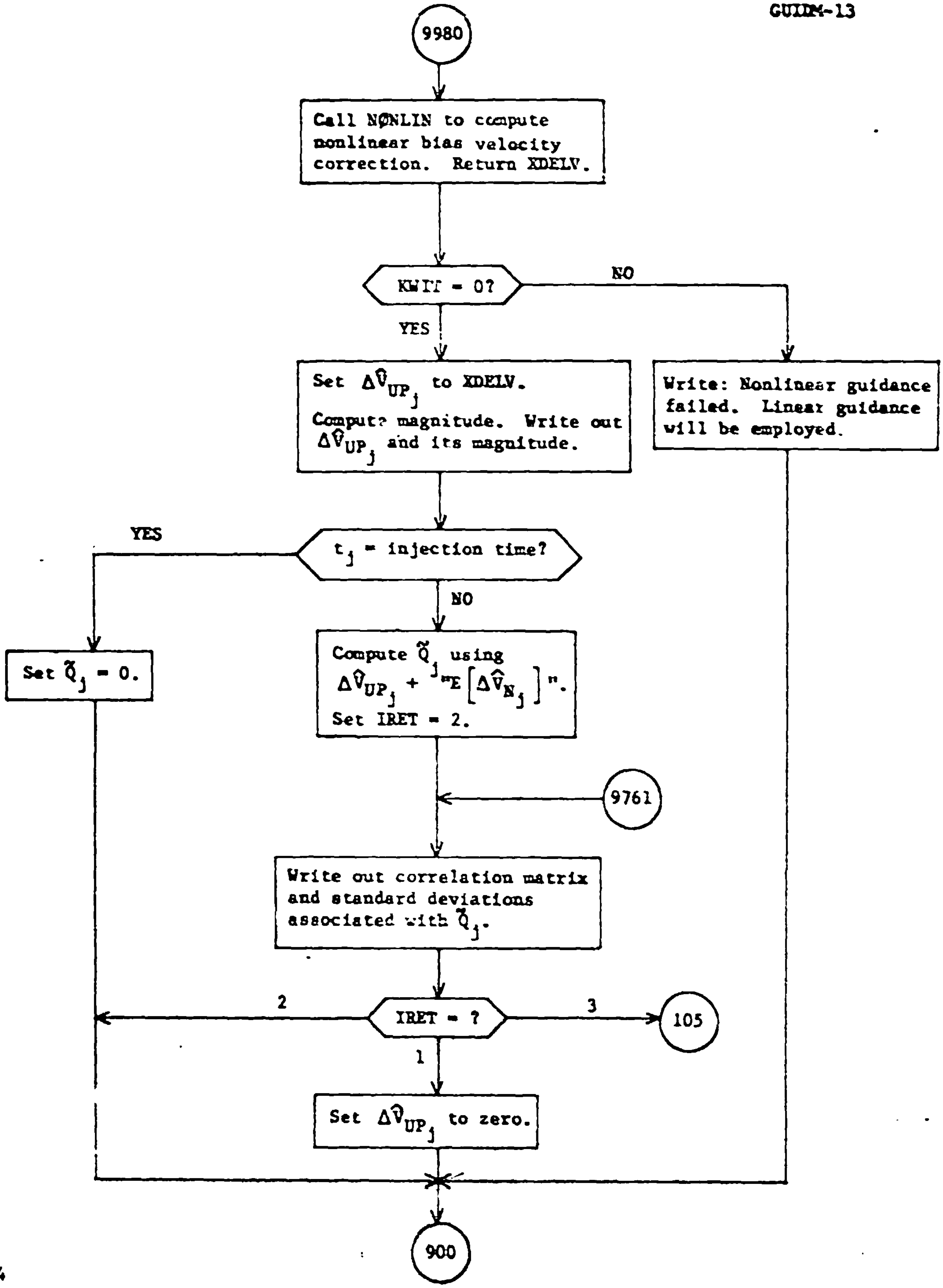


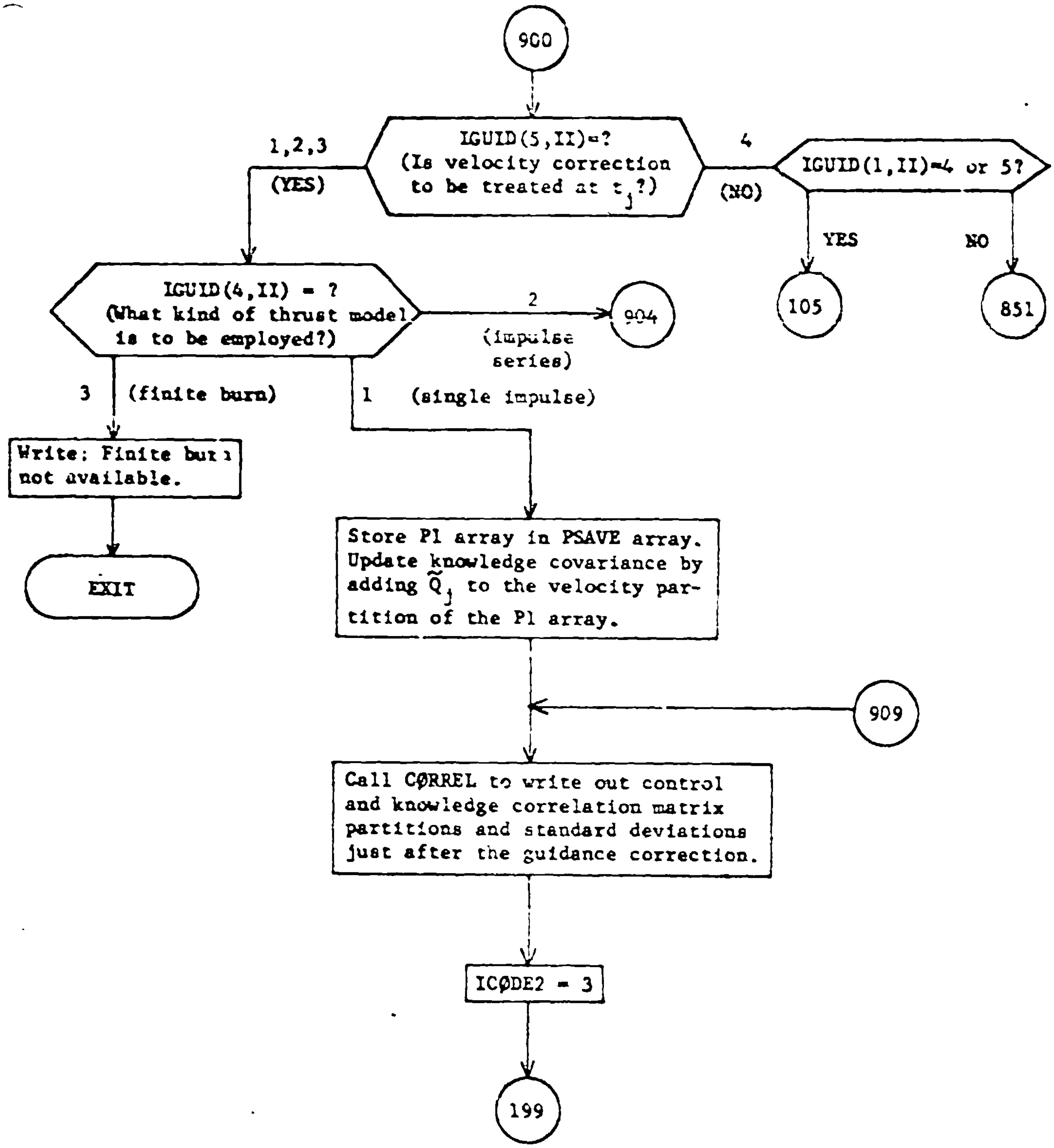

A
Call CØRREL to compute and write out control correlation matrix partitions and standard deviations.
Store position/velocity covariance matrix partition (P1 array) in the PP array for use in subroutine PØICØM.
Set ICØDE2 = 2.
199
Store position partition of the P array in the Z and VEIG arrays.
Call EIGHY to compute and write out the eigenvalues, eigenvectors and hyperellipsoid of the position partition of the P array.
Store velocity partition of the P array in the Z and VEIG arrays.
Call EIGHY to compute and write out the eigenvalues, eigenvectors, and hyperellipsoid of the velocity partition of the P array.
290

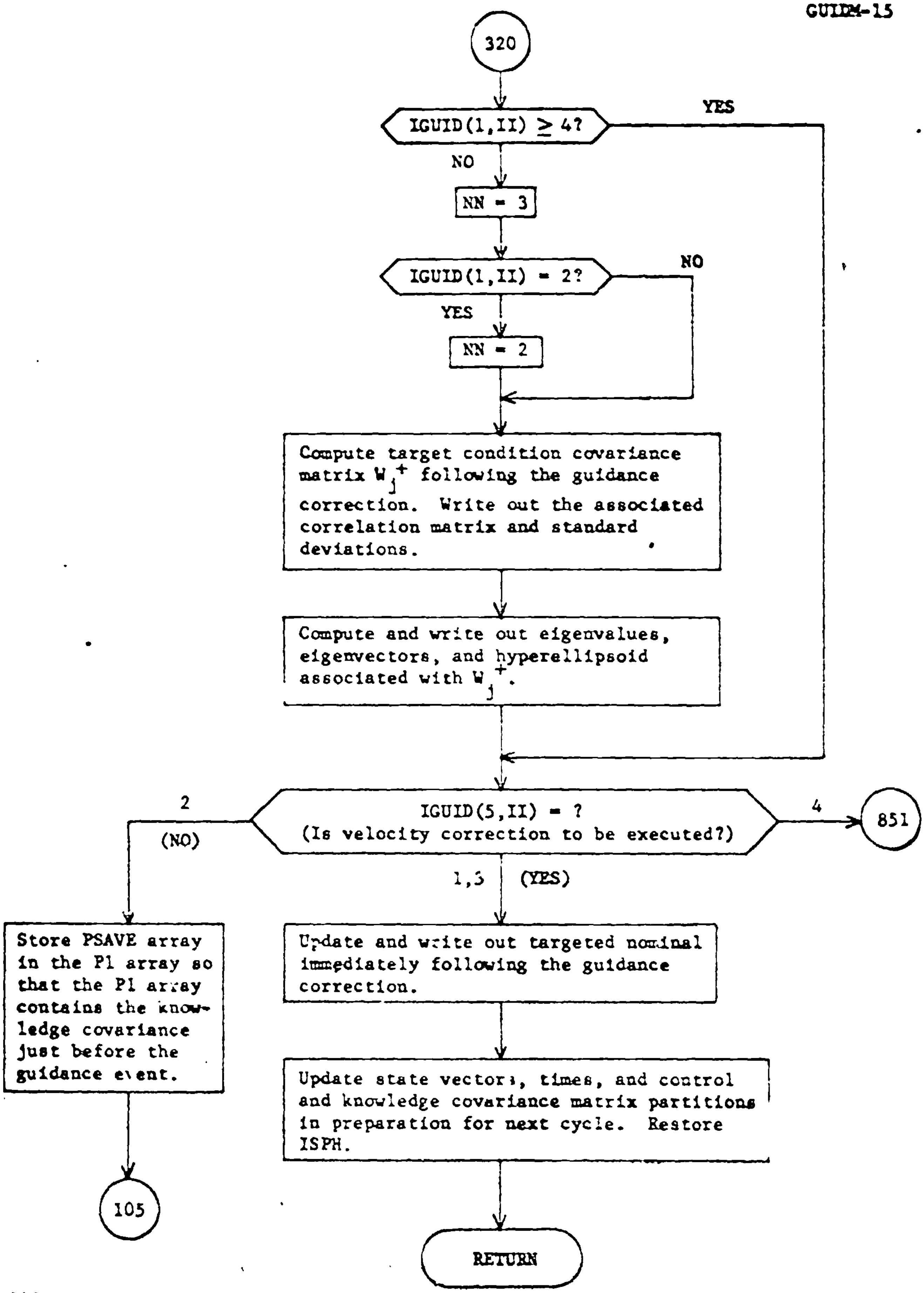

310
Write out complete description of guidance event.
IGUID(5,II) = ?
1
850
2,3,4
IGUID(1,II) = ?
4,5
854
6
851
7
8520
1,2,3
857
Define guidance policy code IGP and execution error code IQP.
Call GUID to compute Γ_j, η_j, W_j^-, and certain quantities required for biased aimpoint guidance.
ISPH = 0?
YES
105
Restore state vector, time, and knowledge covariance matrix partitions to their t_j^- values. Restore ISPH.
RETURN
NO
t_j = injection time?
YES
Set "$E\left[\Delta\hat{V}_{N_j}\right]$" to zero.
862
NO
Compute velocity correction covariance matrix S_j. Write out correlation matrix and standard deviations. Compute and write out eigenvalues and eigenvectors.
IGP = 2?
YES
NO
Compute and write out hyperellipsoid of S_j.
858

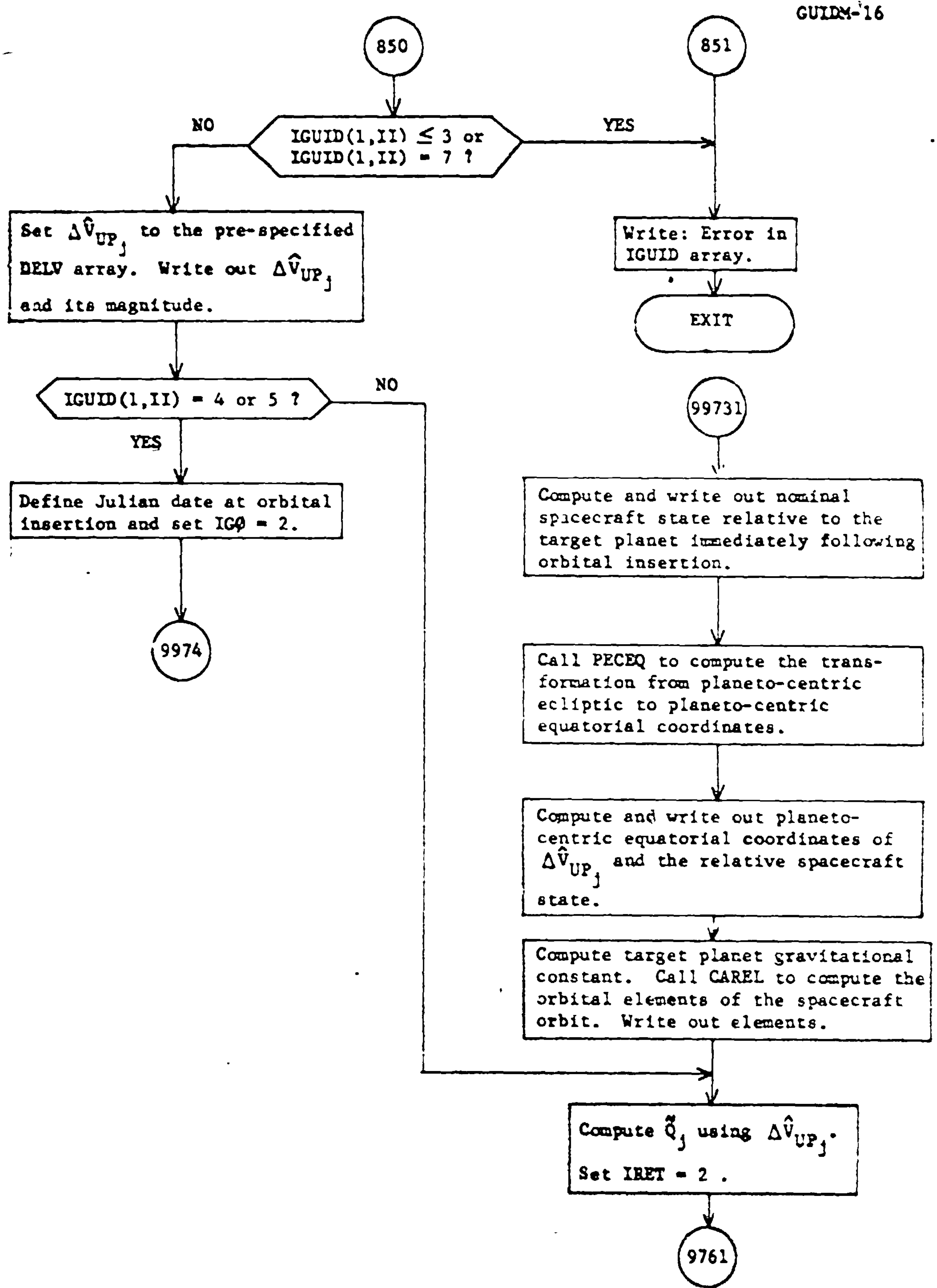

858
IQP = 0?
YES
NO
Use simplified method for computing "E[ΔV̂_Nj]". Write out. Compute and write out Q̃j and its eigenvalues, eigenvectors, and hyperellipsoid.
Use Hoffman-Young formula to compute "E[ΔV̂_Nj]". Write out.
900
862
IGUID(3,II) = 0?
NO
YES
Store "E[ΔV̂_Nj]" in DVN array for use in subroutine BIAIM.
Call BIAIM to perform biased aimpoint guidance event. Return aimpoint μj, bias velocity correction ΔV̂_UPj, and execution error covariance matrix Q̃j.
3
tj = injection time?
NO
YES
Compute Q̃j using "E[ΔV̂_Nj]".
Set IRET = 1
9761
Set Q̃j = 0 and ΔV̂_UPj = 0.
900

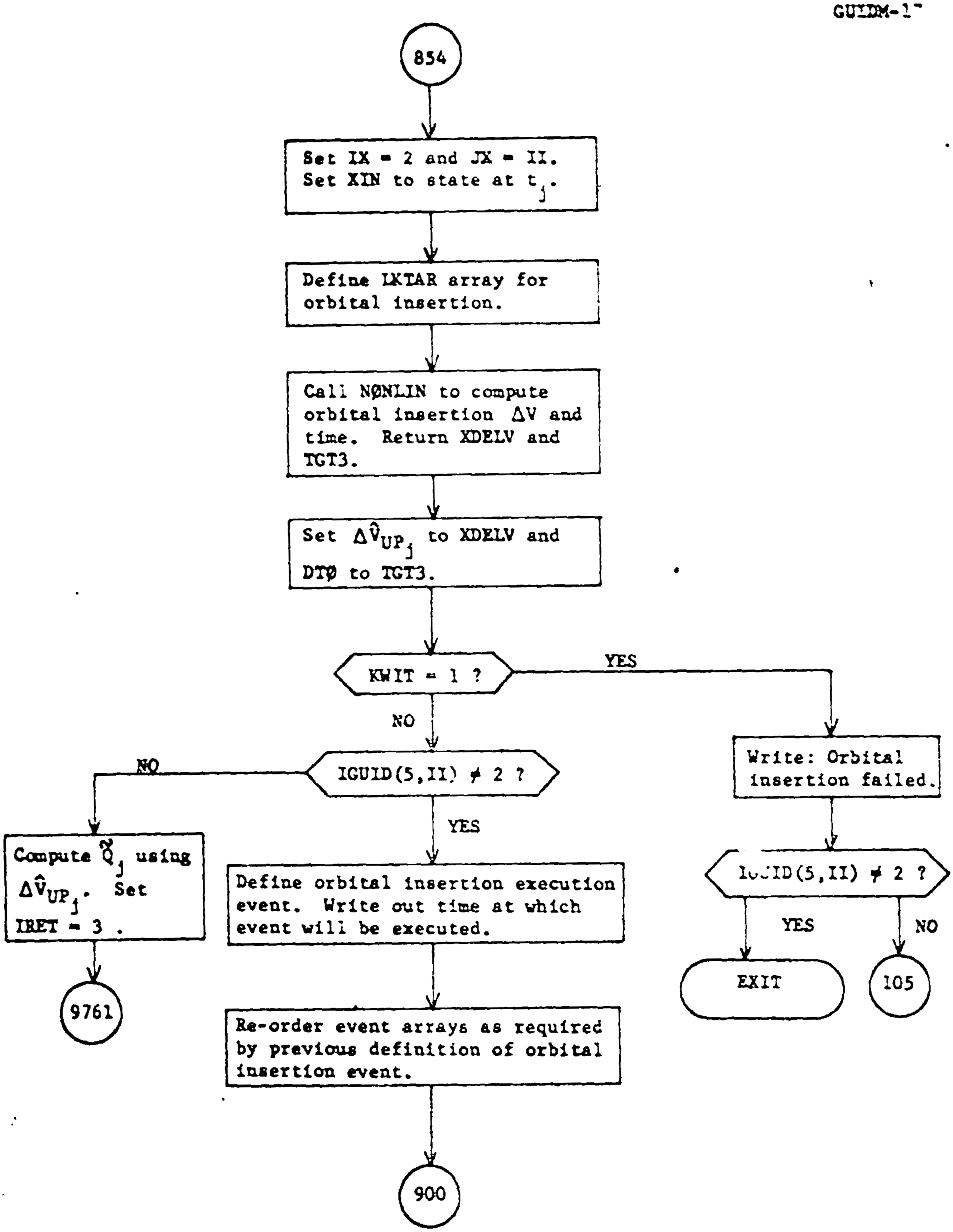

B
IGUID(3,II) = 1?
YES
900
NO
Set IX = 1 and JX = II.
Set XIN to state at t_1.
Write: Bias velocity correction will be recomputed using nonlinear guidance.
IGUID(1,II) = ?
1
FTA
2
2VBP
3
3VBP
Compute Julian date corresponding to t_{CA} and set IGØ = 1.
9974
Compute position and velocity of target planet at the specified time.
IGØ = ?
2
99731
1
Define target variables XTAR, LKTAR, LNPAR, and TGT3 for nonlinear FTA guidance.
Define target variables XTAR, LKTAR, LNPAR, and TGT3 for nonlinear 2VBP guidance.
Define target variables XTAR, LKTAR, LNPAR, and TGT3 for nonlinear 3VBP guidance.
9980

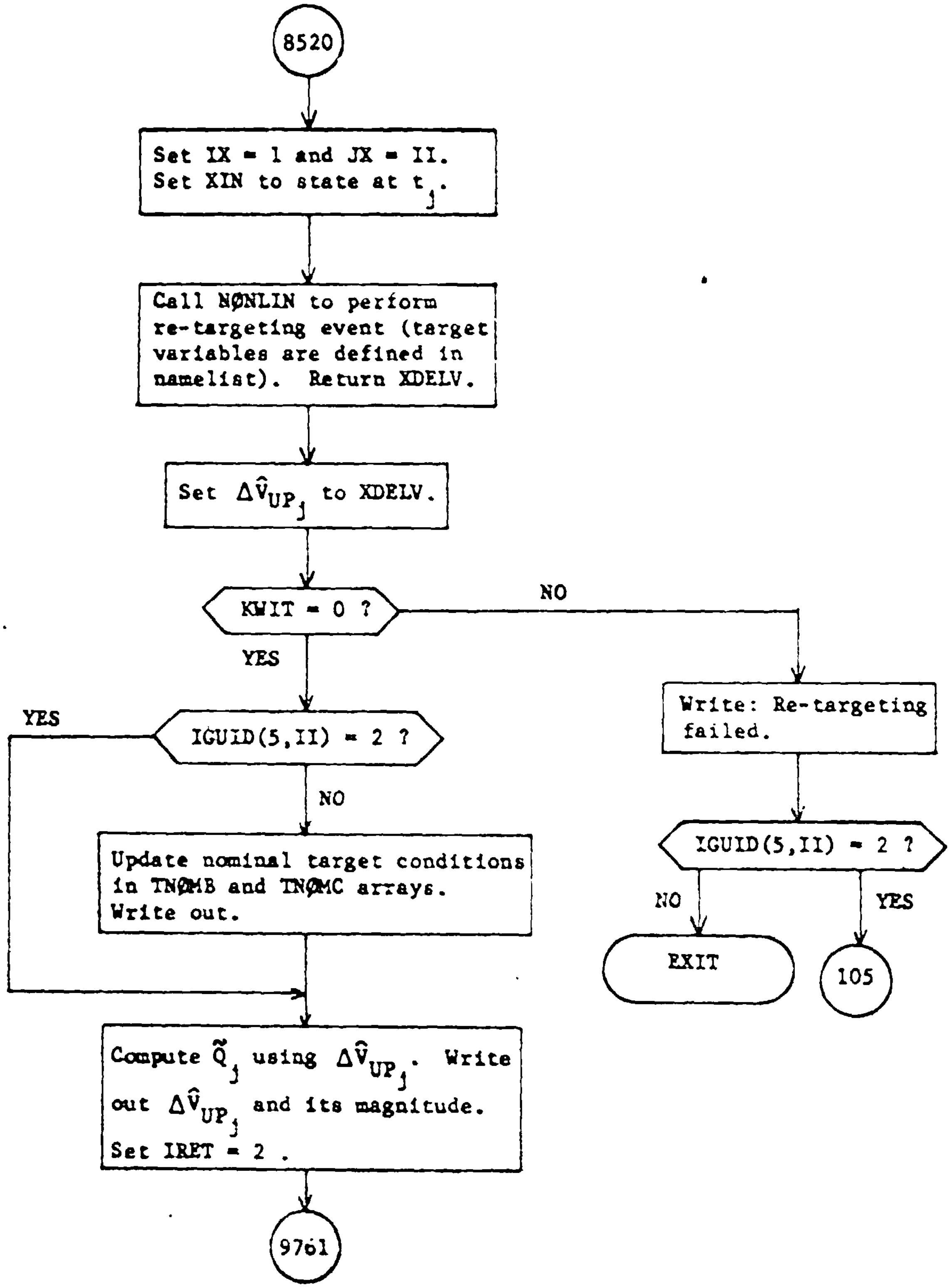

9980
Call NØNLIN to compute nonlinear bias velocity correction. Return XDELV.
KWIT = 0?
NO
YES
Set $\Delta\hat{V}_{UP_j}$ to XDELV.
Compute magnitude. Write out $\Delta\hat{V}_{UP_j}$ and its magnitude.
Write: Nonlinear guidance failed. Linear guidance will be employed.
t_j = injection time?
YES
NO
Set $\tilde{Q}_j = 0$.
Compute $\tilde{Q}_j$ using $\Delta\hat{V}_{UP_j} + {}^{j}"E\left[\Delta\hat{V}_{N_j}\right]"$.
Set IRET = 2.
9761
Write out correlation matrix and standard deviations associated with $\tilde{Q}_j$.
IRET = ?
2
3
105
1
Set $\Delta\hat{V}_{UP_j}$ to zero.
900


900
IGUID(5,II)=?
(Is velocity correction to be treated at t_j?)
1,2,3
(YES)
4
(NO)
IGUID(1,II)=4 or 5?
YES
NO
105
851
IGUID(4,II) = ?
(What kind of thrust model is to be employed?)
2
(impulse series)
904
3
(finite burn)
1
(single impulse)
Write: Finite burn not available.
EXIT
Store P1 array in PSAVE array. Update knowledge covariance by adding $\tilde{Q}_j$ to the velocity partition of the P1 array.
909
Call CØRREL to write out control and knowledge correlation matrix partitions and standard deviations just after the guidance correction.
ICØDE2 = 3
199


320
IGUID(1,II) ≥ 4?
YES
NO
NN = 3
IGUID(1,II) = 2?
NO
YES
NN = 2
Compute target condition covariance matrix W_j^+ following the guidance correction. Write out the associated correlation matrix and standard deviations.
Compute and write out eigenvalues, eigenvectors, and hyperellipsoid associated with W_j^+.
IGUID(5,II) = ?
(Is velocity correction to be executed?)
2
(NO)
4
851
1,3
(YES)
Store PSAVE array in the P1 array so that the P1 array contains the knowledge covariance just before the guidance event.
105
Update and write out targeted nominal immediately following the guidance correction.
Update state vectors, times, and control and knowledge covariance matrix partitions in preparation for next cycle. Restore ISPH.
RETURN


850
851
IGUID(1,II) ≤ 3 or IGUID(1,II) = 7 ?
NO
YES
Set $\Delta\hat{V}_{UP_j}$ to the pre-specified DELV array. Write out $\Delta\hat{V}_{UP_j}$ and its magnitude.
Write: Error in IGUID array.
EXIT
IGUID(1,II) = 4 or 5 ?
NO
YES
Define Julian date at orbital insertion and set IGØ = 2.
9974
99731
Compute and write out nominal spacecraft state relative to the target planet immediately following orbital insertion.
Call PECEQ to compute the transformation from planeto-centric ecliptic to planeto-centric equatorial coordinates.
Compute and write out planeto-centric equatorial coordinates of $\Delta\hat{V}_{UP_j}$ and the relative spacecraft state.
Compute target planet gravitational constant. Call CAREL to compute the orbital elements of the spacecraft orbit. Write out elements.
Compute $\tilde{Q}_j$ using $\Delta\hat{V}_{UP_j}$. Set IRET = 2 .
9761


854
Set IX = 2 and JX = II.
Set XIN to state at t_j.
Define LKTAR array for orbital insertion.
Call NØNLIN to compute orbital insertion ΔV and time. Return XDELV and TGT3.
Set $\Delta\hat{V}_{UP_j}$ to XDELV and DTØ to TGT3.
KWIT = 1 ?
YES
NO
IGUID(5,II) ≠ 2 ?
NO
YES
Compute $\tilde{Q}_j$ using $\Delta\hat{V}_{UP_j}$. Set IRET = 3 .
9761
Define orbital insertion execution event. Write out time at which event will be executed.
Re-order event arrays as required by previous definition of orbital insertion event.
900
Write: Orbital insertion failed.
IGUID(5,II) ≠ 2 ?
YES
NO
EXIT
105


8520
Set IX = 1 and JX = II.
Set XIN to state at t_j.
Call NØNLIN to perform re-targeting event (target variables are defined in namelist). Return XDELV.
Set $\Delta\hat{V}_{UP_j}$ to XDELV.
KWIT = 0 ?
NO
YES
IGUID(5,II) = 2 ?
YES
NO
Update nominal target conditions in TNØMB and TNØMC arrays. Write out.
Compute $\tilde{Q}_j$ using $\Delta\hat{V}_{UP_j}$. Write out $\Delta\hat{V}_{UP_j}$ and its magnitude. Set IRET = 2 .
9761
Write: Re-targeting failed.
IGUID(5,II) = 2 ?
NO
YES
EXIT
105

904

Set XXIN to state at t_j.
Set DELPX to $\Delta\hat{V}_{UP_j}$ + "$\Sigma\left[\Delta\hat{V}_{N_j}\right]$".

Define Julian date at t_j.

Set INPX = 2 and store impulse series execution error variances in the DUMMYQ array.

Call EXCUT to compute the effective execution error covariance matrix corresponding to the impulse series. Return QK.

Restore single impulse execution error variances into the DUMMYQ array.

Store P1 array in the PSAVE array. Update knowledge covariance by adding effective $\tilde{Q}_j$ to the P1 array.

909

PROGRAM GUIS

PURPOSE COMPUTE GUIDANCE MATRIX, VARIATION MATRIX, AND TARGET CONDITION COVARIANCE MATRIX AT A MIDCOURSE GUIDANCE EVENT IN THE SIMULATION PROGRAM

SUBROUTINES SUPPORTED: GUISIM

SUBROUTINES REQUIRED: EPHEM HYELS JACOBI MATIN NTMS ORB PARTL PSIM STMPR VARSIM

LOCAL SYMBOLS

A	TWO-VARIABLE B-PLANE GUIDANCE SUB-MATRIX A
BB	TWO-VARIABLE B-PLANE GUIDANCE SUB-MATRIX B
BDR1	VALUE OF B DOT R RETURNED FROM PARTL (NOT USED)
BDT1	VALUE OF B DOT T RETURNED FROM PARTL (NOT USED)
B1	MAGNITUDE OF B VECTOR RETURNED FROM PARTL (NOT USED)
DUM1	ARRAY OF EIGENVECTORS
DUM	INTERMEDIATE ARRAY
EGVCT	ARRAY OF EIGENVECTORS
EGVL	ARRAY OF EIGENVALUES
ICLS	INTERMEDIATE STORAGE FOR ICL
ICS	INTERMEDIATE STORAGE FOR ICL2
IPR	INTERMEDIATE STORAGE FOR IPRINT
ISPS	INTERMEDIATE STORAGE FOR ISP2
PBR	PARTIAL OF B DOT R WITH RESPECT TO STATE VECTOR
PBT	PARTIAL OF B DOT T WITH RESPECT TO STATE VECTOR
PHI1	INTERMEDIATE ARRAY
PHI2	INTERMEDIATE ARRAY
PHI3	INTERMEDIATE ARRAY

RI1	MOST RECENT NOMINAL SPACECRAFT STATE AT GUIDANCE EVENT
RI	TARGETED NOMINAL SPACECRAFT STATE AT GUIDANCE EVENT
RMCA	SPACECRAFT DISTANCE FROM TARGET PLANET AT CLOSEST APPROACH
RMSI	SPACECRAFT DISTANCE FROM TARGET PLANET AT SPHERE OF INFLUENCE
ROW	TARGET CONDITION CORRELATION MATRIX
RTPS	INERTIAL SPACECRAFT STATE AT SPHERE OF INFLUENCE
SQP	TARGET CONDITION STANDARD DEVIATIONS
TCA	TRAJECTORY TIME AT CLOSEST APPROACH
TSI	TRAJECTORY TIME AT SPHERE OF INFLUENCE
VMCA	MAGNITUDE OF VELOCITY OF VEHICLE RELATIVE TO TARGET PLANET AT CLOSEST APPROACH
VMSI	MAGNITUDE OF VELOCITY OF VEHICLE RELATIVE TO TARGET PLANET AT SPHERE OF INFLUENCE
XCA	SPACECRAFT VELOCITY RELATIVE TO TARGET PLANET AT SPHERE OF INFLUENCE

COMMON COMPUTED/USED:	EM	ICL2	IPRINT	ISPH	ISP2
	NO	XP			
COMMON COMPUTED:	DELTM	TRTM1			
COMMON USED:	ALNGTH	BDR	BDT	B	DATEJ
	DC	DSI	FNTM	FOV	F
	IBARY	IHYP1	ISOI1	ISTMC	NBOD
	NB	NGE	NQE	NTMC	NTP
	ONE	PHI	P	RC	RSI
	TM	VSI	ZERO		

GUIS Analysis

Subroutine GUIS is called at a midcourse guidance event at t_j in the simulation mode to compute three primary quantities for the selected midcourse guidance policy. These three quantities are the variation matrix η_j, the target condition covariance matrix prior to the velocity correction W_j^-, and the guidance matrix Γ_j. Three midcourse guidance policies are available: fixed-time-of-arrival (FTA), two-variable B-plane (2VBP), and three-variable B-plane (3VBP). All are linear impulsive guidance policies having form

$$\Delta\hat{V}_j = \Gamma_j \delta\hat{X}_j$$

where $\Delta\hat{V}_j$ is the commanded velocity correction, and $\delta\hat{X}_j$ is the estimate of the spacecraft position/velocity deviation from the targeted nominal. The relevant equations for each guidance policy will be summarized below.

The variation matrix η_j for FTA guidance relates deviations in spacecraft state at t_j to position deviations at time of closest approach t_{CA}, and is given by

$$\eta_j = \left[\phi_1 \;\vdots\; \phi_2\right]$$

where $\left[\phi_1 \;\vdots\; \phi_2\right]$ is the upper half of the state transition matrix $\Phi(t_{CA},t_j)$. The guidance matrix for FTA guidance is given by

$$\Gamma_j = \left[-\phi_2^{-1}\phi_1 \;\vdots\; -I\right] .$$

The variation matrix for 3VBP guidance relates deviations in spacecraft state at t_j to deviations in B·T, B·R, and t_{SI}, where t_{SI} is the time at which the sphere of influence is pierced. Unlike the variation matrix for FTA guidance, which can be computed analytically or by numerical differencing, the 3VBP variation matrix must always be computed using numerical differencing since no good analytical formulas are available which relate deviations in spacecraft state at t_j to deviations in t_{SI}. If the variation matrix is written as

$$\eta_j = \left[\eta_1 \;\vdots\; \eta_2\right]$$

then the guidance matrix for 3VBP guidance is given by

$$\Gamma_j = \left[-\eta_2^{-1}\eta_1 \;\vdots\; -I\right] .$$

The variation matrix for 2VBP guidance relates deviations in spacecraft state at t_j to deviations in B·T and B·R and is given by

$$\eta_j = M\Phi(t_{SI}, t_j)$$

where M is an analytically computed matrix relating B·T and B·R deviations to spacecraft state deviations at t_{SI}, and $\Phi(t_{SI}, t_j)$ is the state transition matrix over $\left[t_j, t_{SI}\right]$. If η_j is written as

$$\eta_j = \left[A \mid B\right]$$

then the guidance matrix for 2VBP guidance is given by

$$\Gamma_j = \left[-B^T(BB^T)^{-1}A \mid -B^T(BB^T)^{-1}B\right]$$

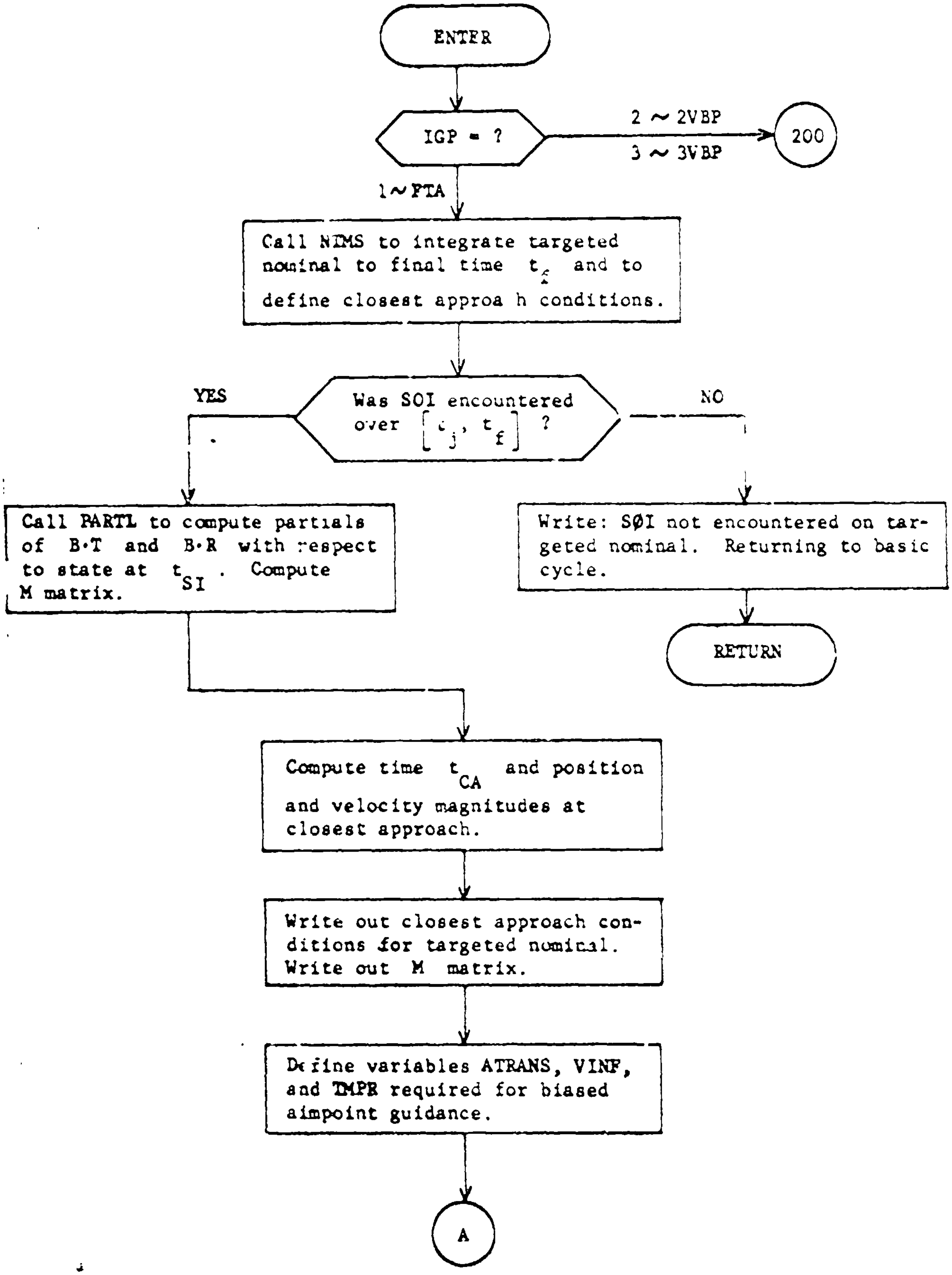
All state transition matrices and, hence, all variation matrices used by the above three guidance policies are referenced to the most recent nominal trajectory for improved numerical accuracy.

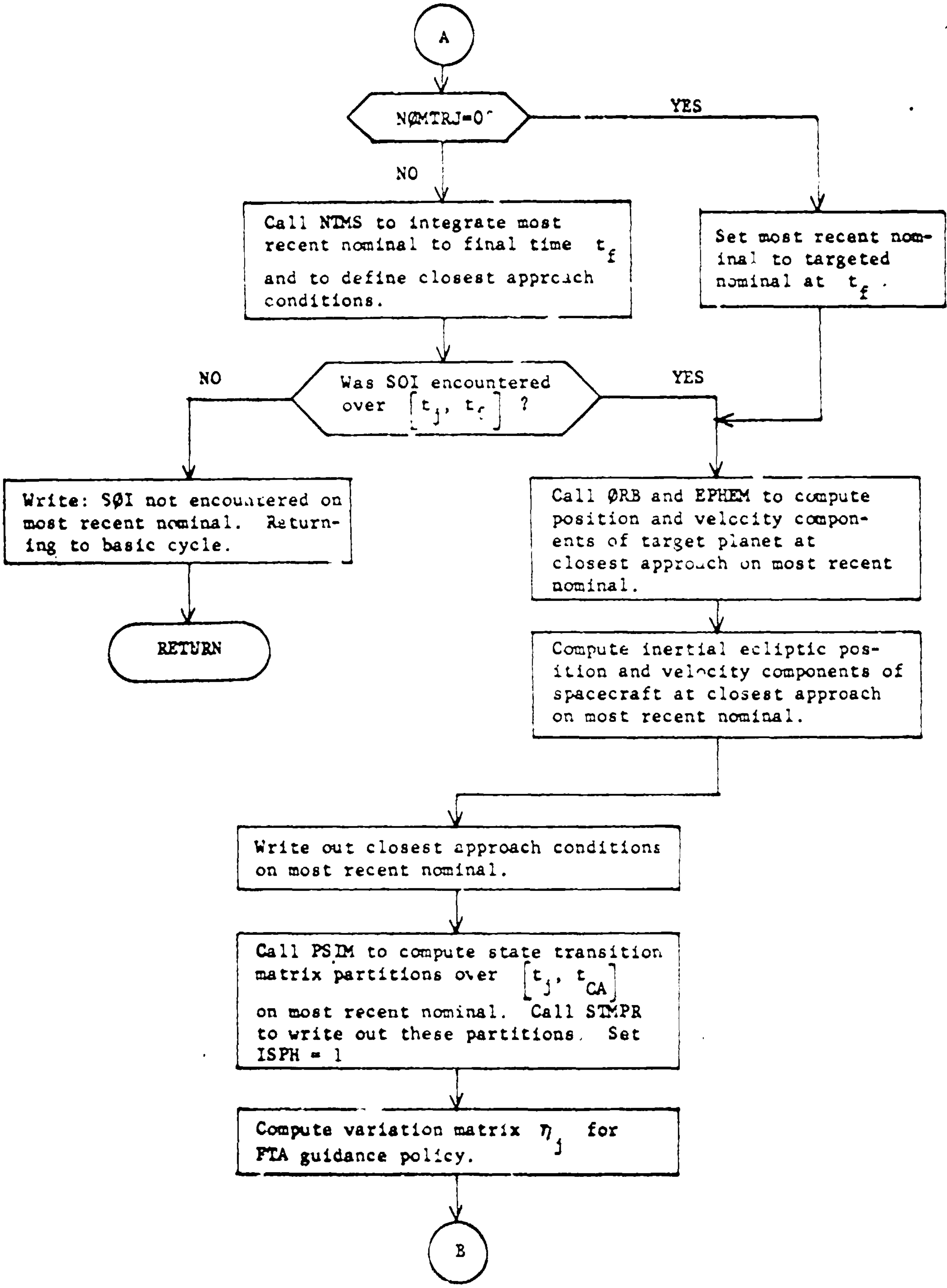
Once the variation matrix η_j is available for any of the above guidance policies, the target condition covariance matrix can be computed using

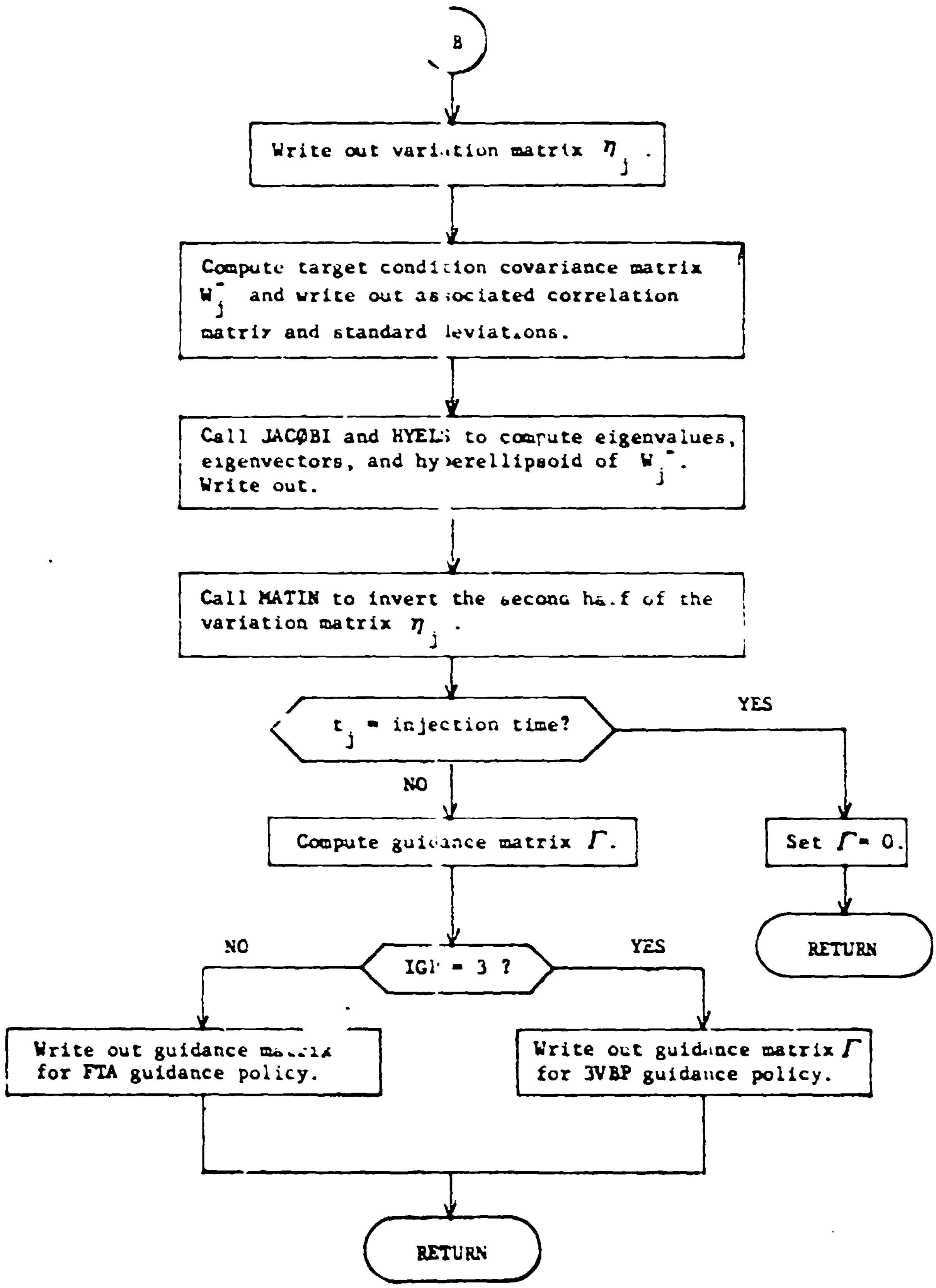
$$W_j^- = \eta_j P_{c_j}^- \eta_j^T$$

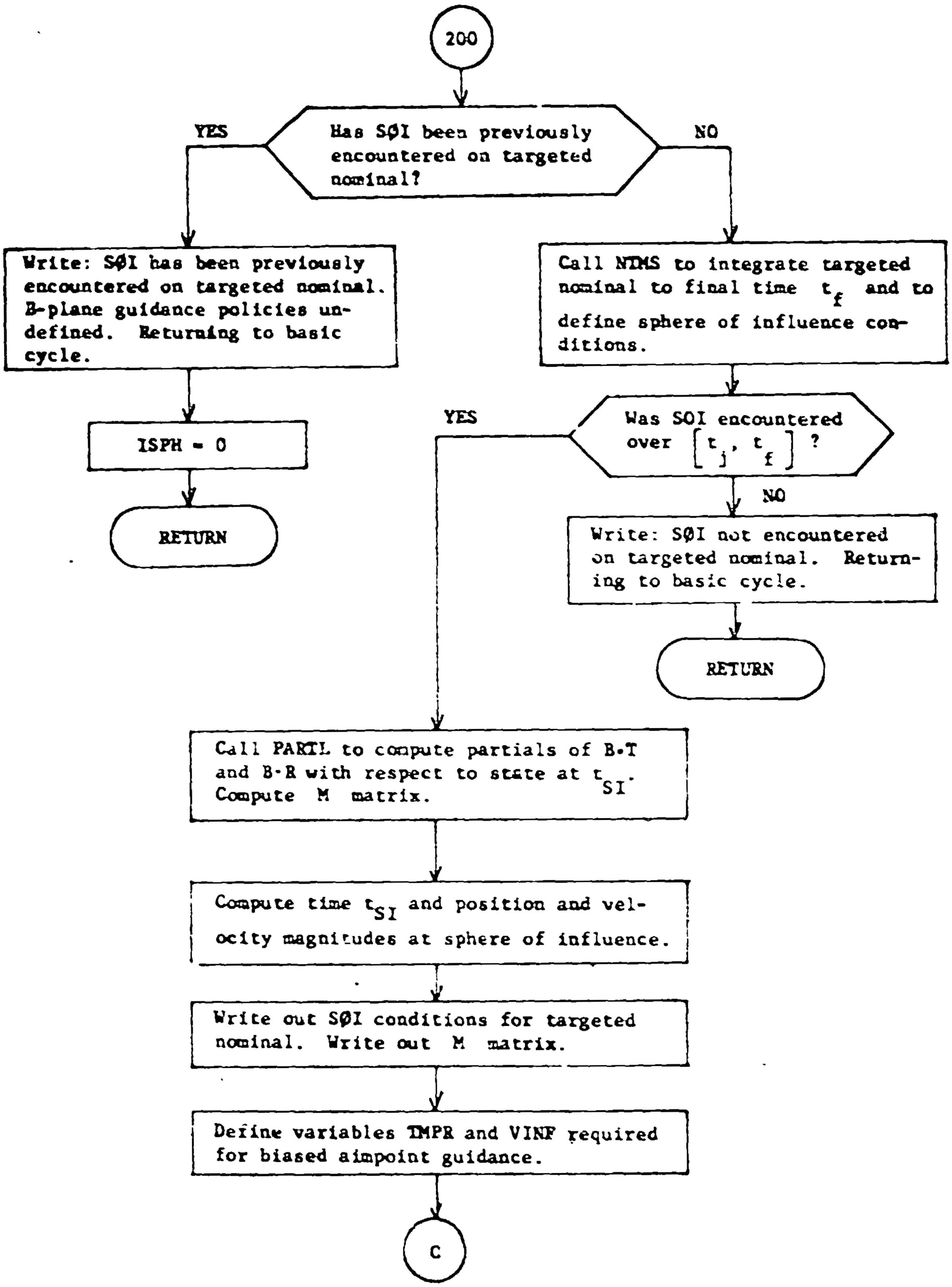
where $P_{c_j}^-$ is the control covariance matrix immediately prior to the guidance event.

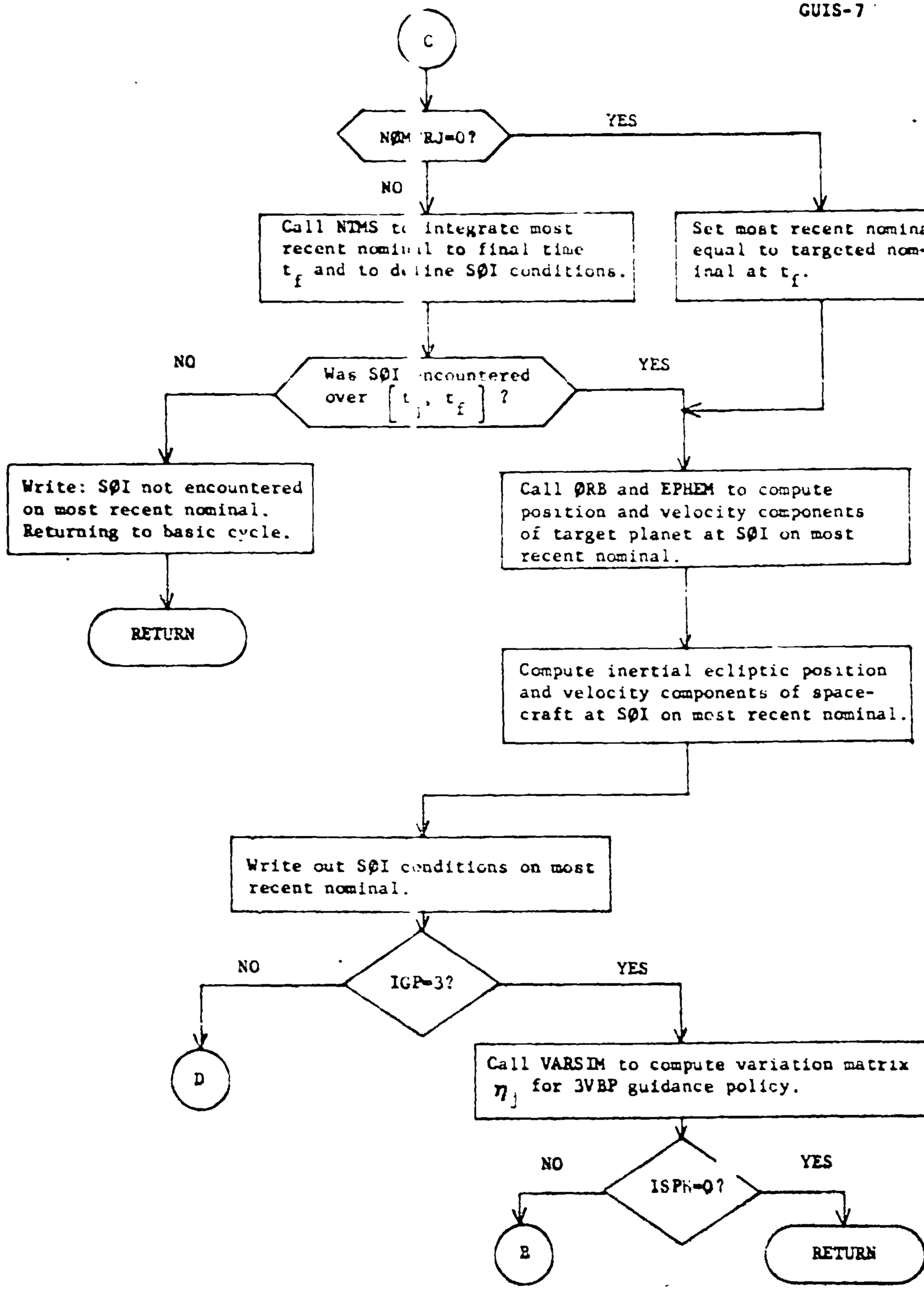
GUIS Flow Chart

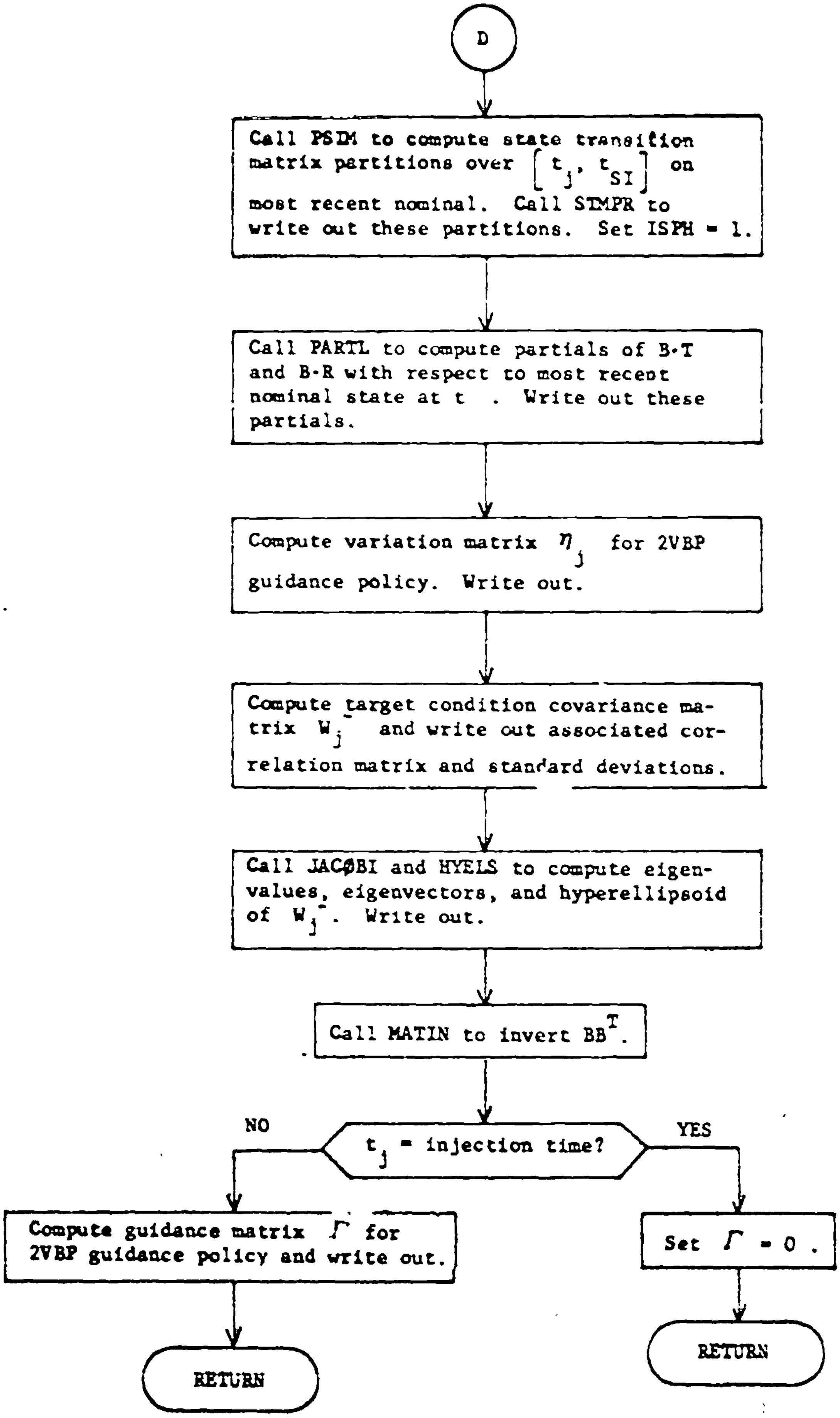



A
NØMTRJ=0?
YES
NO
Call NTMS to integrate most recent nominal to final time t_f and to define closest approach conditions.
Set most recent nominal to targeted nominal at t_f.
Was SOI encountered over $[t_j, t_f]$?
NO
YES
Write: SØI not encountered on most recent nominal. Returning to basic cycle.
RETURN
Call ØRB and EPHEM to compute position and velocity components of target planet at closest approach on most recent nominal.
Compute inertial ecliptic position and velocity components of spacecraft at closest approach on most recent nominal.
Write out closest approach conditions on most recent nominal.
Call PSIM to compute state transition matrix partitions over $[t_j, t_{CA}]$ on most recent nominal. Call STMPR to write out these partitions. Set ISPH = 1
Compute variation matrix η_j for FTA guidance policy.
B


B
Write out variation matrix η_j.
Compute target condition covariance matrix W_j^- and write out associated correlation matrix and standard deviations.
Call JACØBI and HYELS to compute eigenvalues, eigenvectors, and hyperellipsoid of W_j^-. Write out.
Call MATIN to invert the second half of the variation matrix η_j.
t_j = injection time?
YES
NO
Compute guidance matrix Γ.
Set $\Gamma = 0$.
RETURN
NO
IGP = 3 ?
YES
Write out guidance matrix for FTA guidance policy.
Write out guidance matrix Γ for 3VBP guidance policy.
RETURN


200
Has SØI been previously encountered on targeted nominal?
YES
NO
Write: SØI has been previously encountered on targeted nominal. B-plane guidance policies undefined. Returning to basic cycle.
ISPH = 0
RETURN
Call NTMS to integrate targeted nominal to final time t_f and to define sphere of influence conditions.
Was SOI encountered over $[t_j, t_f]$?
YES
NO
Write: SØI not encountered on targeted nominal. Returning to basic cycle.
RETURN
Call PARTL to compute partials of B·T and B·R with respect to state at t_{SI}. Compute M matrix.
Compute time t_{SI} and position and velocity magnitudes at sphere of influence.
Write out SØI conditions for targeted nominal. Write out M matrix.
Define variables TMPR and VINF required for biased aimpoint guidance.
C


C
NØM RJ=0?
YES
NO
Call NTMS to integrate most recent nominal to final time t_f and to define SØI conditions.
Set most recent nominal equal to targeted nominal at t_f.
NO
Was SØI encountered over $[t_j, t_f]$?
YES
Write: SØI not encountered on most recent nominal. Returning to basic cycle.
Call ØRB and EPHEM to compute position and velocity components of target planet at SØI on most recent nominal.
RETURN
Compute inertial ecliptic position and velocity components of spacecraft at SØI on most recent nominal.
Write out SØI conditions on most recent nominal.
NO
IGP=3?
YES
D
Call VARSIM to compute variation matrix η_j for 3VBP guidance policy.
NO
ISPK=0?
YES
B
RETURN


D
Call PSIM to compute state transition matrix partitions over [t_j, t_SI] on most recent nominal. Call STMPR to write out these partitions. Set ISPH = 1.
Call PARTL to compute partials of B·T and B·R with respect to most recent nominal state at t . Write out these partials.
Compute variation matrix η_j for 2VBP guidance policy. Write out.
Compute target condition covariance matrix W_j^- and write out associated correlation matrix and standard deviations.
Call JACØBI and HYELS to compute eigenvalues, eigenvectors, and hyperellipsoid of W_j^-. Write out.
Call MATIN to invert BB^T.
t_j = injection time?
NO
YES
Compute guidance matrix Γ for 2VBP guidance policy and write out.
Set Γ = 0 .
RETURN
RETURN

PROGRAM GUISIM

PURPOSE CONTROL EXECUTION OF GUIDANCE EVENT IN THE SIMULATION PROGRAM

SUBROUTINES SUPPORTED: SIMULL

SUBROUTINES REQUIRED: CORREL DYNOS GUIS HYELS JACOBI NAVM PSIM STMPR

LOCAL SYMBOLS:

ADA	VARIATION MATRIX
AK1	ACTUAL RESOLUTION ERROR
AL1	ACTUAL ERROR IN POINTING ANGLE ALPHA
BT1	ACTUAL ERROR IN POINTING ANGLE BETA
CXSU1	STORAGE FOR CXSU KNOWLEDGE COVARIANCE
CXSV1	STORAGE FOR CXSV KNOWLEDGE COVARIANCE
CXU1	STORAGE FOR CXU KNOWLEDGE COVARIANCE
CXV1	STORAGE FOR CXV KNOWLEDGE COVARIANCE
CXXS1	STORAGE FOR CXXS KNOWLEDGE COVARIANCE
DELX	ESTIMATED STATE DEVIATION FROM TARGETED NOMINAL TRAJECTORY
DUM1	INTERMEDIATE VARIABLE
DUM2	ARRAY OF EIGENVECTORS
DVCM	MAGNITUDE OF COMMANDED MIDCOURSE VELOCITY CORRECTION
DVC	COMMANDED MIDCOURSE VELOCITY CORRECTION
DVE	ERROR IN MIDCOURSE VELOCITY CORRECTION DUE TO NAVIGATION UNCERTAINTY
DV	PERFECT MIDCOURSE VELOCITY CORRECTION
DX	ACTUAL STATE DEVIATION FROM TARGETED NOMINAL TRAJECTORY
EGVCT	ARRAY OF EIGENVECTORS
EGVL	ARRAY OF EIGENVALUES
EXEC	EXECUTION ERROR COVARIANCE MATRIX

EXM	MAGNITUDE OF UPDATE VELOCITY CORRECTION
GAP	INTERMEDIATE ARRAY EQUAL TO GA TIMES P
GA	GUIDANCE MATRIX
ICODE2	INTERNAL CONTROL FLAG
IGP	MIDCOURSE GUIDANCE POLICY CODE
OUT	SPACECRAFT VELOCITY RELATIVE TO TARGET PLANET IN PLANETO-CENTRIC EQUATORIAL COORDINATES
PS1	STORAGE FOR PS KNOWLEDGE COVARIANCE
P1	STORAGE FOR P KNOWLEDGE COVARIANCE
RF1	MOST RECENT NOMINAL SPACECRAFT STATE AT GUIDANCE EVENT
RF	TARGETED NOMINAL SPACECRAFT STATE AT GUIDANCE EVENT
ROW	INTERMEDIATE VECTOR
SQP	INTERMEDIATE VECTOR
S1	ACTUAL PROPORTIONALITY ERROR
VEIG	MATRIX TO BE DIAGONALIZED
Z	INTERMEDIATE ARRAY

COMMON COMPUTED/USED:	ADEVX	CXSUG	CXSU	CXSVG	CXSV
	CXUG	CXU	CXVG	CXV	CXXSG
	CXXS	EDEVX	ICODE	NGE	PG
	PSG	PS	P	RI1	TG
	XF1	XG			
COMMON COMPUTED:	DELTM	TRTM1	XI1	XI	
COMMON USED:	AALP	ABET	ADEVXS	APRO	ARES
	EDEVXS	FOP	FOV	ICOT3	IEIG
	IHYP1	ISPH	ISTMC	NDIM1	NDIM2
	NDIM3	Q	SIGALP	SIGBET	SIGPRO
	SIGRES	TEVN	U0	V0	W
	XF	XSL	ZERO		

GUISIM Analysis

Subroutine GUISIM is the executive guidance subroutine in the simulation program. In addition to controlling the computational flow for all types of guidance events, GUISIM also performs many of the required guidance computations itself.

Before considering each type of guidance event, the treatment of a general guidance event will be discussed. Let t_j be the time at which the guidance event occurs. Before any guidance event can be executed the targeted nominal state $\tilde{X}_j$, most recent nominal state X_j^-, estimated state deviation $\delta\hat{X}_j^-$ from most recent nominal, actual state deviation $\delta\widetilde{X}_j^-$ from most recent nominal, knowledge covariance $P_{K_j}^-$, and control covariance $P_{c_j}^-$ must all be available, where $( )^-$ indicates values immediately before the event. Only the control covariance is not available prior to entering GUISIM. The propagation of the control covariance over the interval $[t_{j-1}, t_j]$, where t_{j-1} denotes the time of the previous guidance event, is performed within GUISIM.

The next step in the treatment of a general guidance event is concerned with the computation of the commanded velocity correction, execution error covariance, actual execution error, and actual velocity correction. In the simulation program a non-statistical commanded velocity correction can always be computed. This commanded velocity correction $\Delta\hat{V}_j$ is used to compute the execution error covariance matrix $\widetilde{Q}_j$ and the actual execution error $\delta\Delta V_j$. A summary of the execution error model and the equations used to compute $\widetilde{Q}_j$ and $\delta\Delta V_j$ can be found in the subroutine QCØMP analysis section. The actual velocity correction is then computed using the equation

$$\Delta V_j = \Delta\hat{V}_j + \delta\Delta V_j$$

The last step is concerned with the updating of required quantities prior to returning to the basic cycle. An assumption underlying the modeled guidance process is that the targeted nominal is always updated by the commanded velocity correction. In the simulation program the update velocity correction $\Delta\hat{V}_{UP_j}$ is always identical to the commanded velocity correction $\Delta\hat{V}_j$. This is in contrast to the error analysis program where $\Delta\hat{V}_{UP_j}$ is equated with the non-statistical component of $\Delta\hat{V}_j$. The

most recent and targeted nominal states immediately following the guidance event are updated using the equations

$$\tilde{X}_j^+ = \tilde{X}_j^- + \delta\hat{\tilde{X}}_j^- + \begin{bmatrix} 0 \\ \delta\Delta\hat{\bar{V}}_{UP_j} \end{bmatrix}$$

$$\bar{X}_j^+ = \tilde{X}_j^+$$

The actual and estimated state deviations from the most recent nominal are given by

$$\delta\tilde{X}_j^+ = \delta\tilde{X}_j^- - \delta\hat{\tilde{X}}_j^- + \begin{bmatrix} 0 \\ \delta\tilde{\Delta V}_j \end{bmatrix}$$

$$\delta\hat{\tilde{X}}_j^+ = 0$$

The previous 4 equations assume an impulsive thrust model. If, instead, the thrust is modeled as an impulse series, then an effective estimated state $\hat{X}_{eff}$ and an effective actual state X_{eff} are computed. The equations used to compute these effective states are summarized in the subroutine PULSEX analysis section. The previous update equations are then replaced by the following equations

$$\tilde{X}_j^+ = \hat{X}_{eff}$$

$$\bar{X}_j^+ = \tilde{X}_j^+$$

$$\delta\tilde{X}_j^+ = X_{eff} - \hat{X}_{eff}$$

$$\delta\hat{X}_j^+ = 0$$

The knowledge covariance is updated using the equation

$$P_{K_j}^+ = P_{K_j}^- + \left[\begin{array}{c|c} 0 & 0 \\ \hline 0 & \tilde{Q}_j \end{array}\right]$$

if an impulsive thrust model is assumed. If the thrust is modeled as a series of impulses, then an effective execution error covariance $\tilde{Q}_{eff}$ is computed and the knowledge covariance is updated using the equation

$$P^{+}_{K_j} = P^{-}_{K_j} + \tilde{Q}_{eff}$$

In either case the control covariance is updated simply by setting

$$P^{+}_{c_j} = P^{+}_{K_j}$$

This equation is a direct consequence of the assumption that the targeted nominal is always updated at a guidance event.

A "compute only" option is available in GUISIM in which all of the $(\ )^{+}$ quantities will still be computed and printed. However, states, deviations, and covariances are then reset to their former $(\ )^{-}$ values prior to returning to the basic cycle.

Each specific type of guidance event involves the computation of other quantities not discussed above. These will be covered in the following discussion of specific guidance events.

1. Midcourse and biased aimpoint guidance.

Linear midcourse guidance policies have form

$$\Delta\hat{V}_{N_j} = \Gamma_j \, \delta\hat{X}_j$$

where the subscript N indicates that this is the velocity correction required to null out deviations from the nominal target state. This notation is required to differentiate between this type of velocity correction and velocity corrections required to achieve an altered target state. Linear midcourse guidance policies are discussed in more detail in the subroutine GUIS analysis section.

Subroutine GUISIM calls GUIS to compute the guidance matrix, Γ_j, and the target condition covariance immediately prior to the guidance event, W^{-}_j, and then uses Γ_j to compute the velocity correction covariance S_j, which is defined as

$$S_j = E\,[\Delta\hat{V}_{N_j}\,\Delta\hat{V}^T_{N_j}],$$

and is given by the equation

$$S_j = \Gamma_j (P^-_{c_j} - P^-_{K_j})\ \Gamma_j^T$$

This equation assumes that an optimal estimation algorithm is employed in the navigation process, since the derivation of this equation requires the orthogonality of the estimate and the estimation error.

Since state estimates $\delta\hat{X}_j$ are generated in the simulation program, an actual $\Delta\hat{V}_{N_j}$ can always be computed. This is in contrast to the error analysis program where only a statistical or effective $\Delta\hat{V}_N$ can be computed. The perfect velocity correction $\underline{\Delta V}_j$, defined as the velocity correction required to null out actual deviations from the nominal target state, is also computed for midcourse guidance events. Assuming linear guidance theory, the perfect velocity correction is given by

$$\underline{\Delta V}_j = \Gamma_j\ \delta X_j$$

where δX_j is the actual deviation from the targeted nominal. An option is also available in GUISIM for re-computing $\Delta\hat{V}_{N_j}$ using nonlinear techniques. However, it should be noted that the nonlinear two-variable B-plane guidance policy, unlike the corresponding linear policy, constrains the z-component of ΔV_{N_j} to be zero.

If planetary quarantine constraints must be satisfied at a midcourse correction, GUISIM calls BIAIM to compute the new aimpoint μ_j and the bias velocity correction $\Delta\hat{V}_{B_j}$. All computations in BIAIM are based on linear guidance theory. However, an option is available in GUISIM to recompute the total velocity correction $\Delta\hat{V}_{B_j} + \Delta\hat{V}_{N_j}$, but not μ_j, using nonlinear techniques. This option is recommended if a biased aimpoint guidance event occurs at t_j = injection time. It should also be noted that Q_j is set to zero if t_j = injection time since it is assumed that the injection

covariance does not change for small changes in injection velocity.

After the updated control covariance $P_{c_j}^{+}$ has been computed, the target condition covariance matrix W_j^{+} following the guidance correction is computed using the equation

$$W_j^{+} = \eta_j \, P_{c_j}^{+} \, \eta_j^{T}$$

where variation matrix η_j has been previously computed in subroutine GUIS.

2. Re-targeting.

In the simulation (and error analysis) program a re-targeting event is defined to be the computation of a velocity correction $\Delta \hat{V}_{RT}$ required to achieve a new set of target conditions using nonlinear techniques. Since the state estimate $\tilde{X}_j^{-} + \delta\tilde{\hat{X}}_j^{-}$ is used as the zero-th iterate in the re-targeting process, the new target conditions must be close enough to the original nominal target conditions to ensure a convergent process.

It should be noted that after a re-targeting event the new target conditions are henceforth treated as the nominal target conditions.

3. Orbital insertion.

An orbital insertion event is divided into a decision event and an execution event. At a decision event the orbital insertion velocity correction $\Delta\hat{V}_{\phi I}$ and the time interval Δt separating decision and execution are computed based on the state estimate $\tilde{X}_j^{-} + \delta\tilde{\hat{X}}_j^{-}$. The relevant equations can be found in the subroutine CØPINS analysis section for coplanar orbital insertion; in NØPINS, for non-planar orbital insertion. Before returning to the basic cycle, GUISIM schedules the orbital insertion execution event to occur at $t_j + \Delta t$ and re-orders the necessary event arrays accordingly.

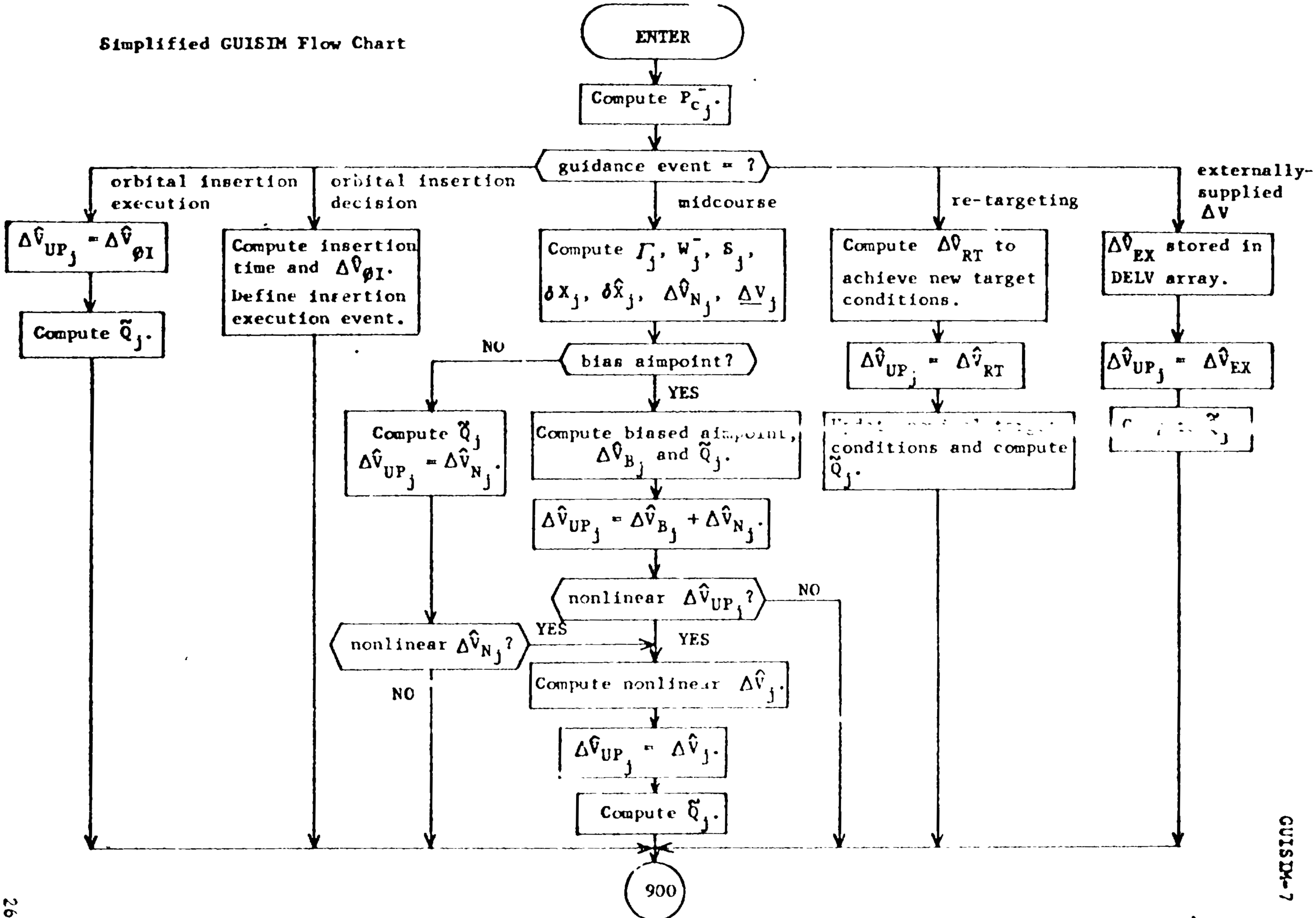
At an orbital insertion execution event the previously computed $\Delta\hat{V}_{\phi I}$ is used to update the targeted nominal state. In addition, the planeto-centric equatorial components of $\Delta\hat{V}_{\phi I}$ and the actual spacecraft cartesian and orbital element states following the insertion maneuver are computed.

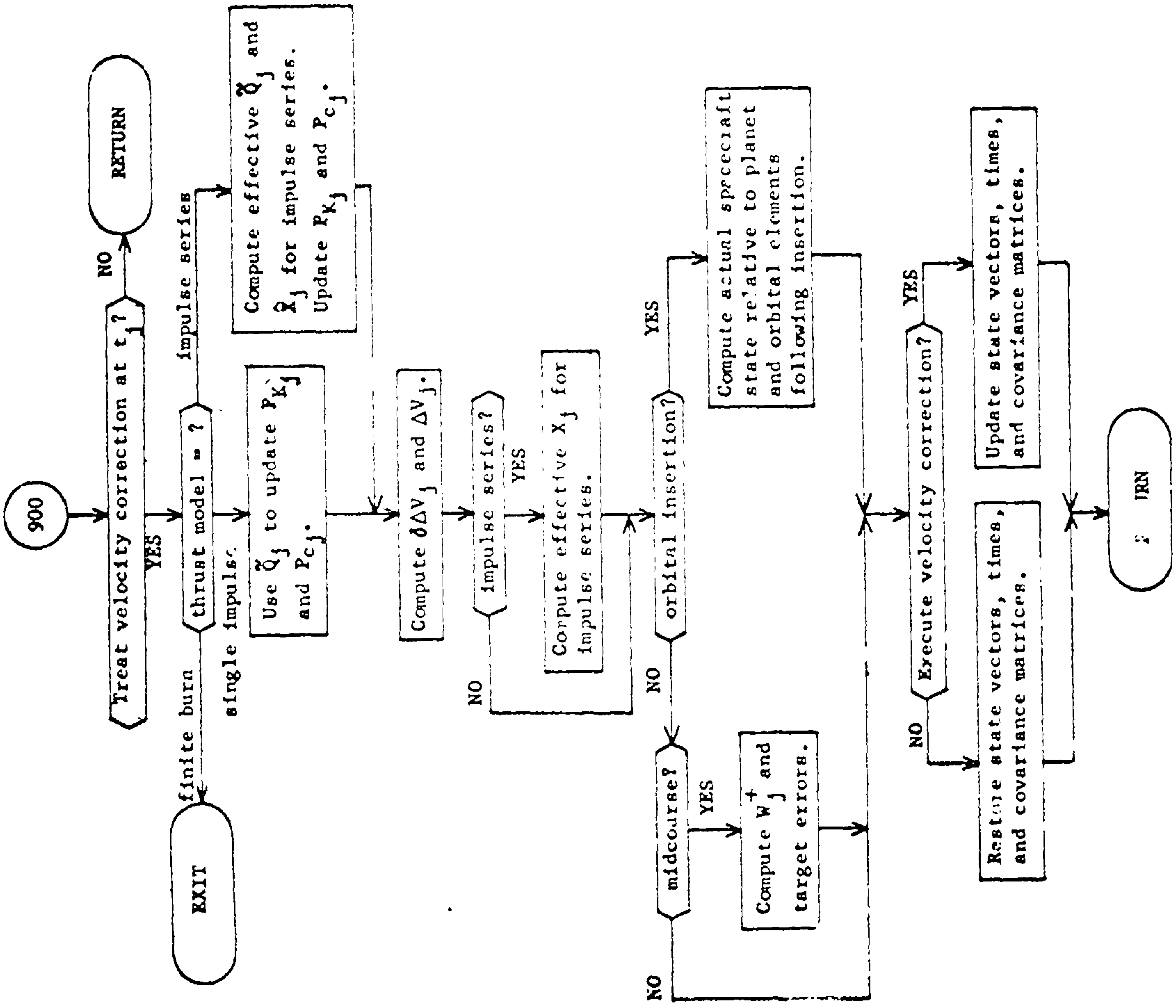
4. **Externally-supplied velocity correction.**

At this type of guidance event the state estimate $\tilde{X}_j^- + \tilde{\hat{X}}_j^-$ is simply updated using the externally-supplied velocity correction $\Delta\hat{V}_{EX}$.

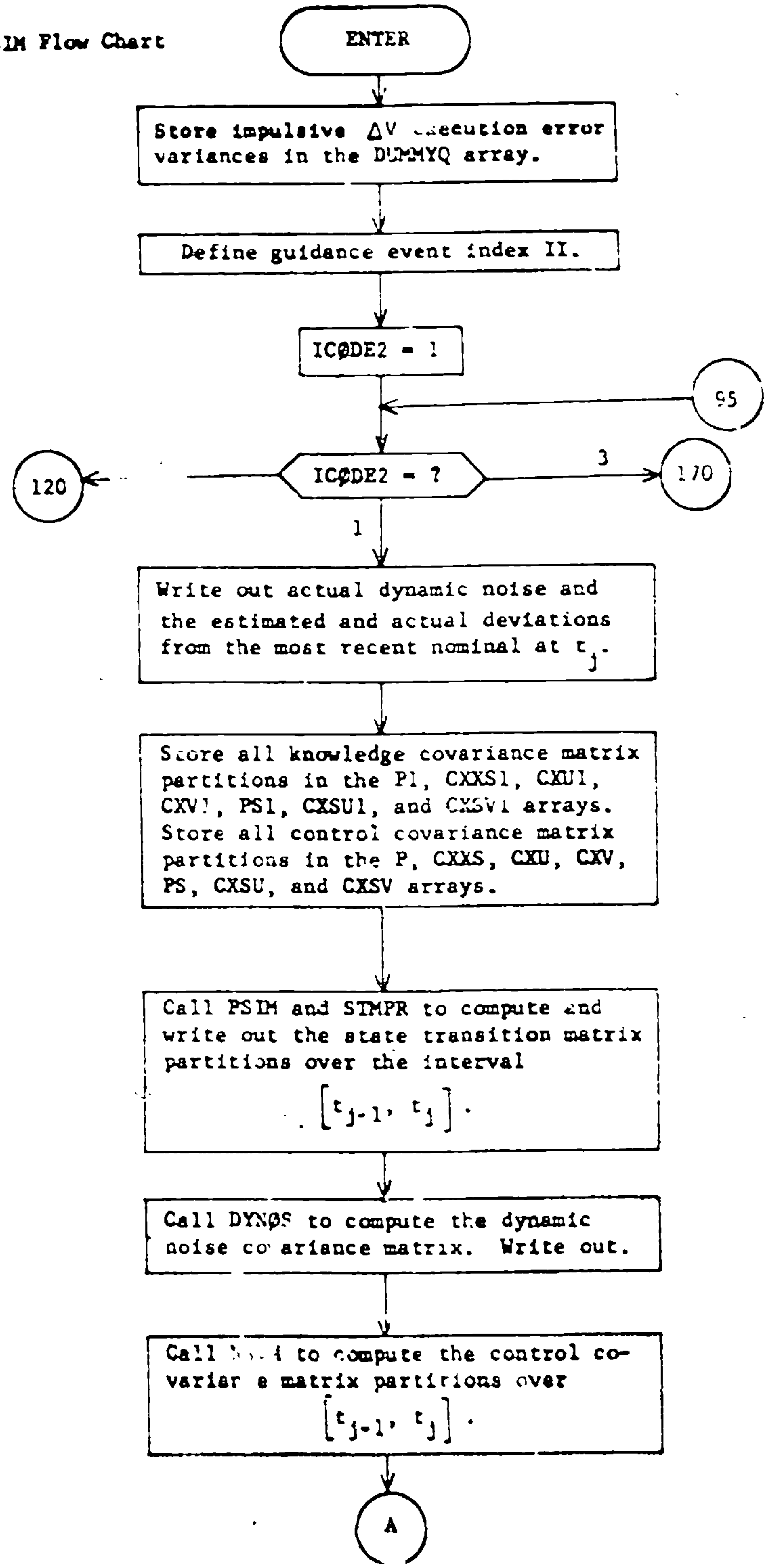
Because of the complexity of the GUISIM flow chart, a simplified flow chart depicting the main elements of the GUISIM structure precedes the complete GUISIM flow chart.

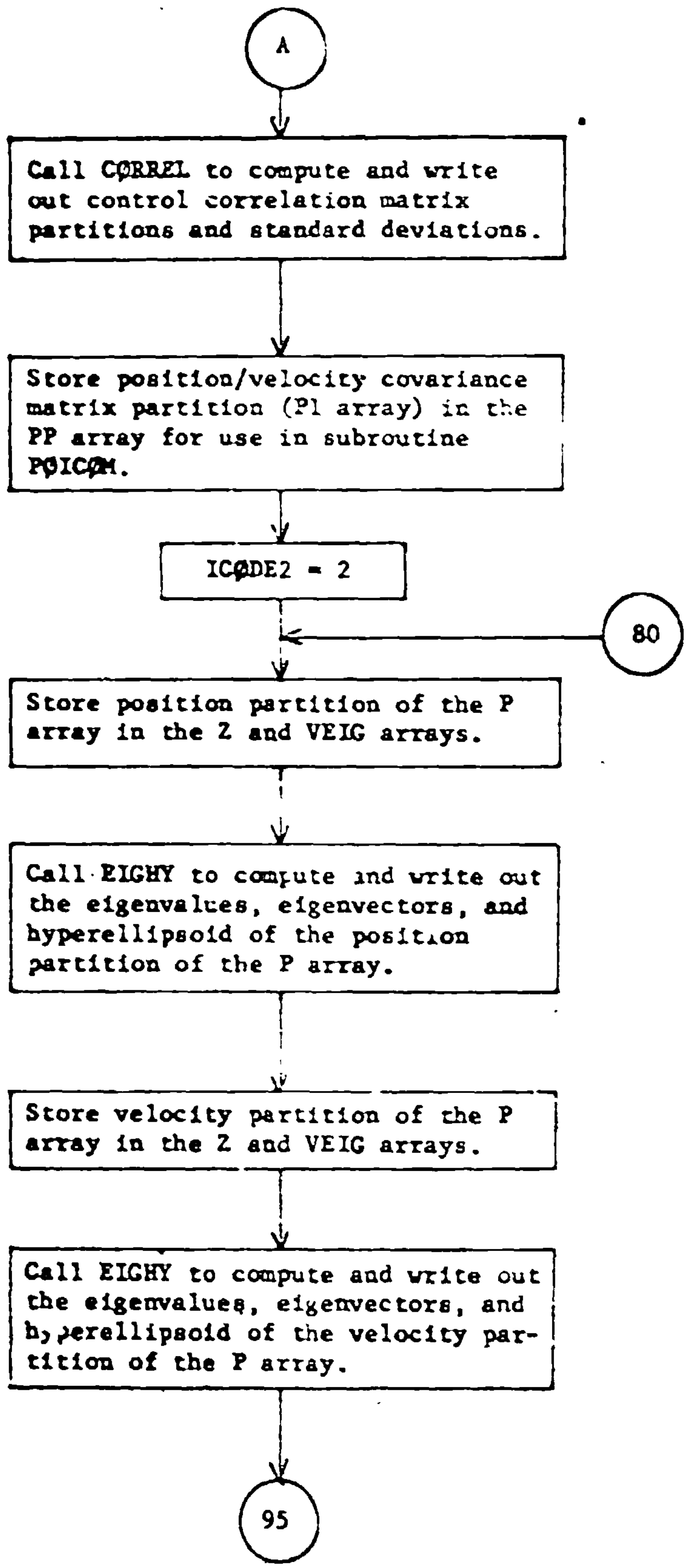
Simplified GUISIM Flow Chart

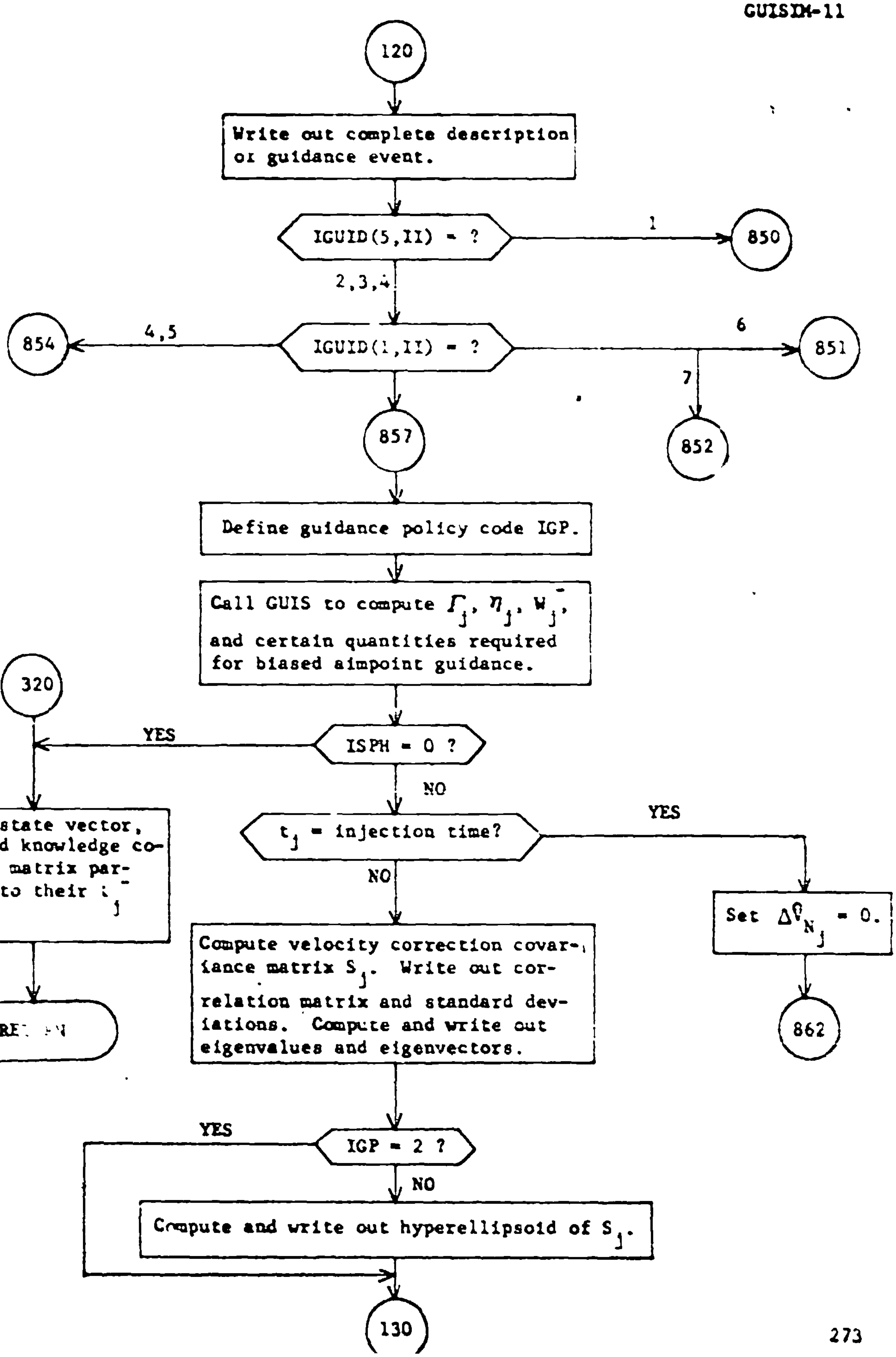


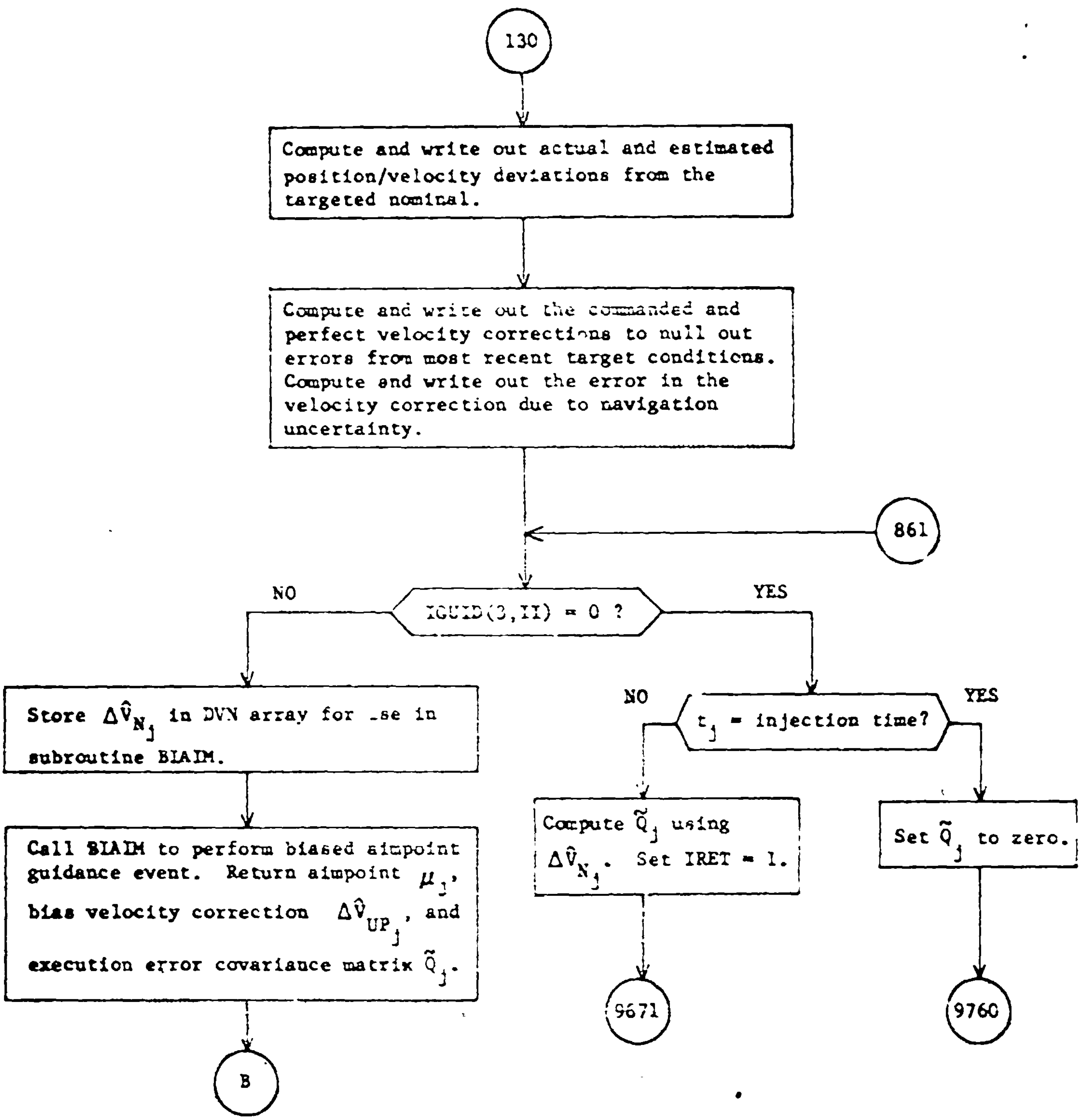

900
Treat velocity correction at t_j?
NO
RETURN
YES
thrust model = ?
impulse series
Compute effective $\tilde{Q}_j$ and $\tilde{X}_j$ for impulse series. Update P_{K_j} and P_{c_j}.
single impulse
Use $\tilde{Q}_j$ to update P_{K_j} and P_{c_j}.
finite burn
EXIT
Compute $\delta\Delta V_j$ and ΔV_j.
impulse series?
YES
Compute effective X_j for impulse series.
NO
orbital insertion?
YES
Compute actual spacecraft state relative to planet and orbital elements following insertion.
NO
midcourse?
YES
Compute W_j^+ and target errors.
NO
Execute velocity correction?
YES
Update state vectors, times, and covariance matrices.
NO
Restore state vectors, times, and covariance matrices.

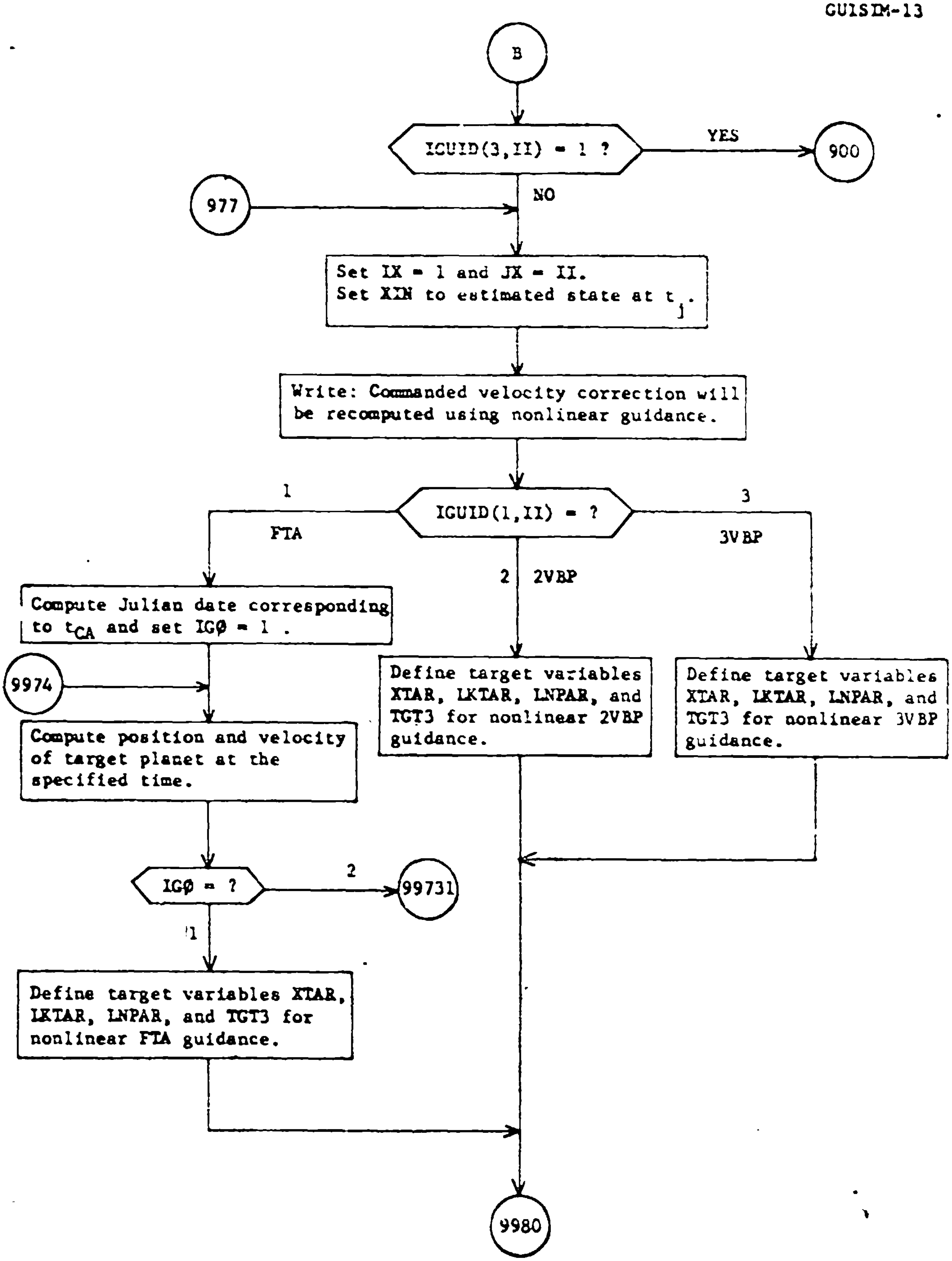
GUISIM Flow Chart

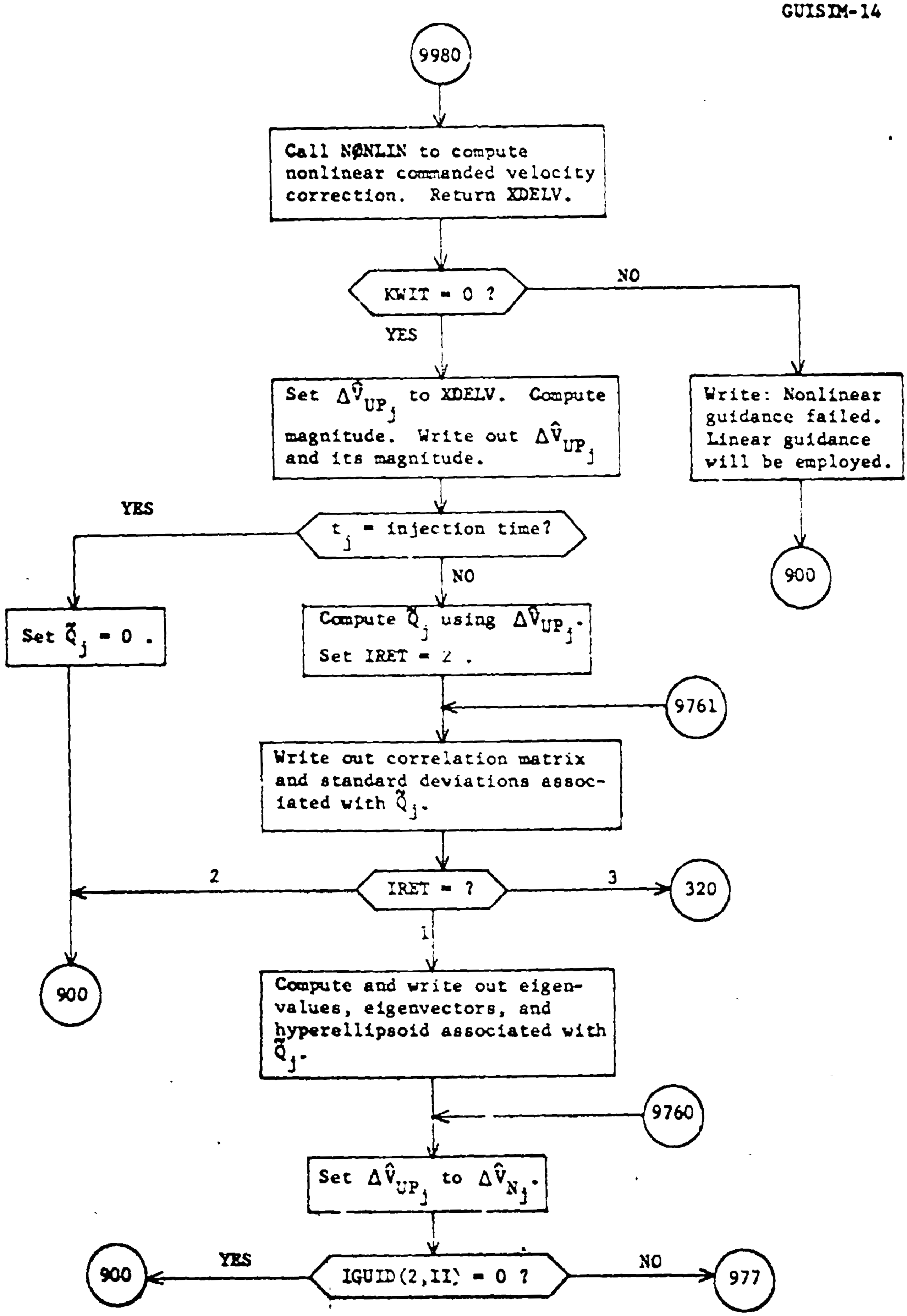


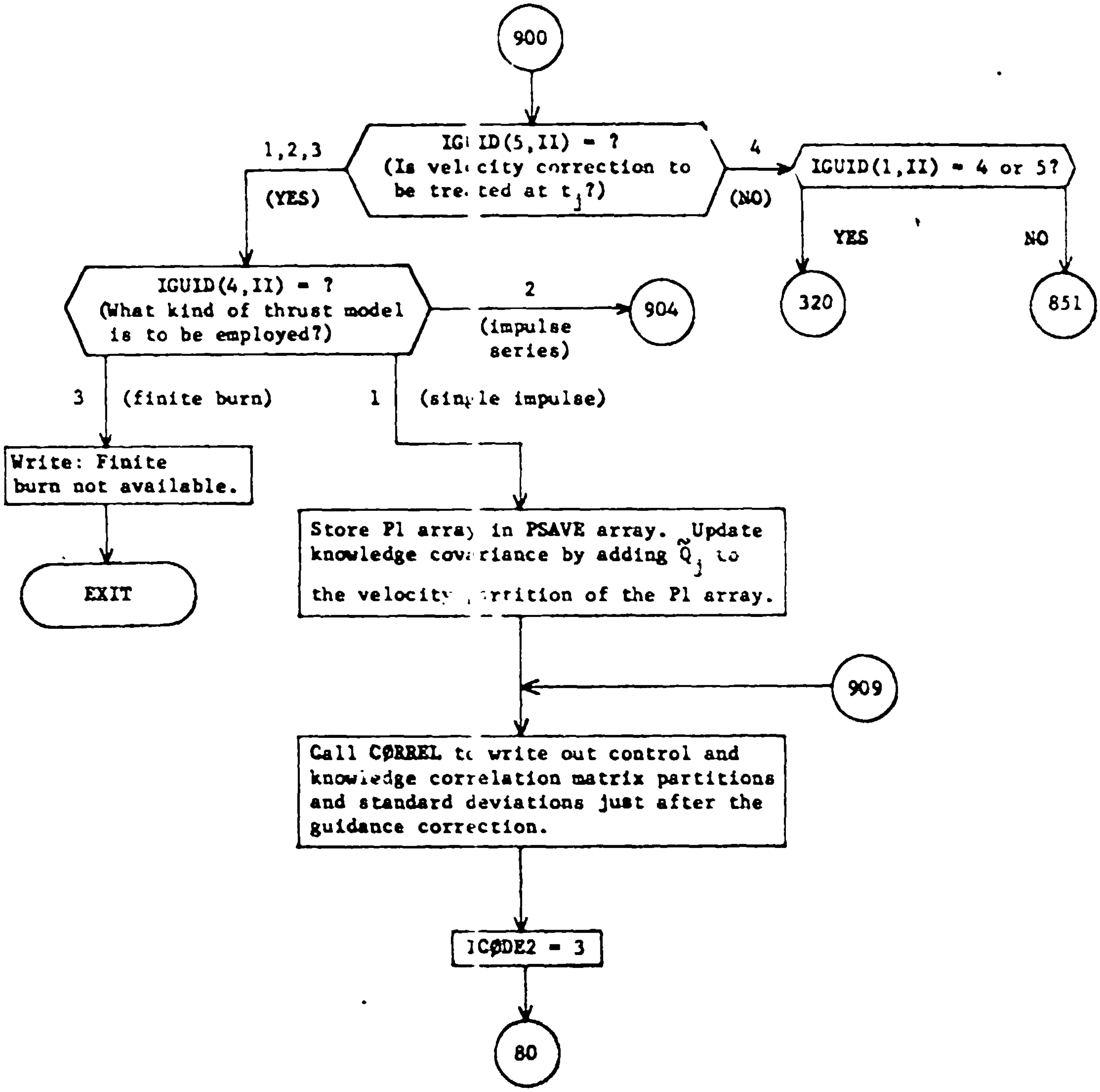

A
Call CØRREL to compute and write out control correlation matrix partitions and standard deviations.
Store position/velocity covariance matrix partition (P1 array) in the PP array for use in subroutine PØICØM.
ICØDE2 = 2
80
Store position partition of the P array in the Z and VEIG arrays.
Call EIGHY to compute and write out the eigenvalues, eigenvectors, and hyperellipsoid of the position partition of the P array.
Store velocity partition of the P array in the Z and VEIG arrays.
Call EIGHY to compute and write out the eigenvalues, eigenvectors, and hyperellipsoid of the velocity partition of the P array.
95

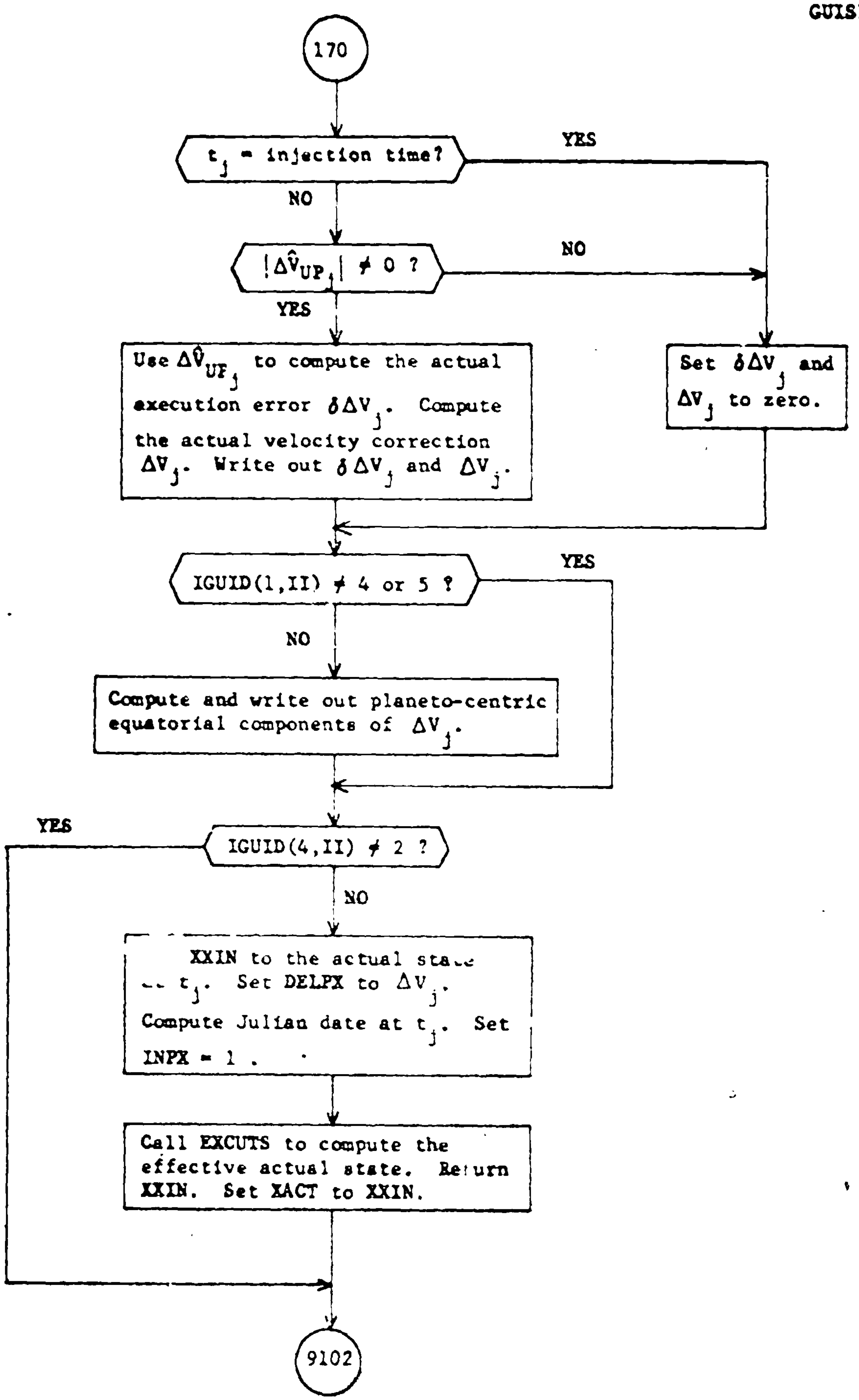

120
Write out complete description of guidance event.
IGUID(5,II) = ?
1
850
2,3,4
IGUID(1,II) = ?
4,5
854
6
851
7
852
857
Define guidance policy code IGP.
Call GUIS to compute Γ_j, η_j, W_j^-, and certain quantities required for biased aimpoint guidance.
320
ISPH = 0 ?
YES
NO
Restore state vector, time, and knowledge covariance matrix partitions to their t_j^- values.
RETURN
t_j = injection time?
YES
NO
Set $\Delta\hat{V}_{N_j} = 0$.
862
Compute velocity correction covariance matrix S_j. Write out correlation matrix and standard deviations. Compute and write out eigenvalues and eigenvectors.
IGP = 2 ?
YES
NO
Compute and write out hyperellipsoid of S_j.
130

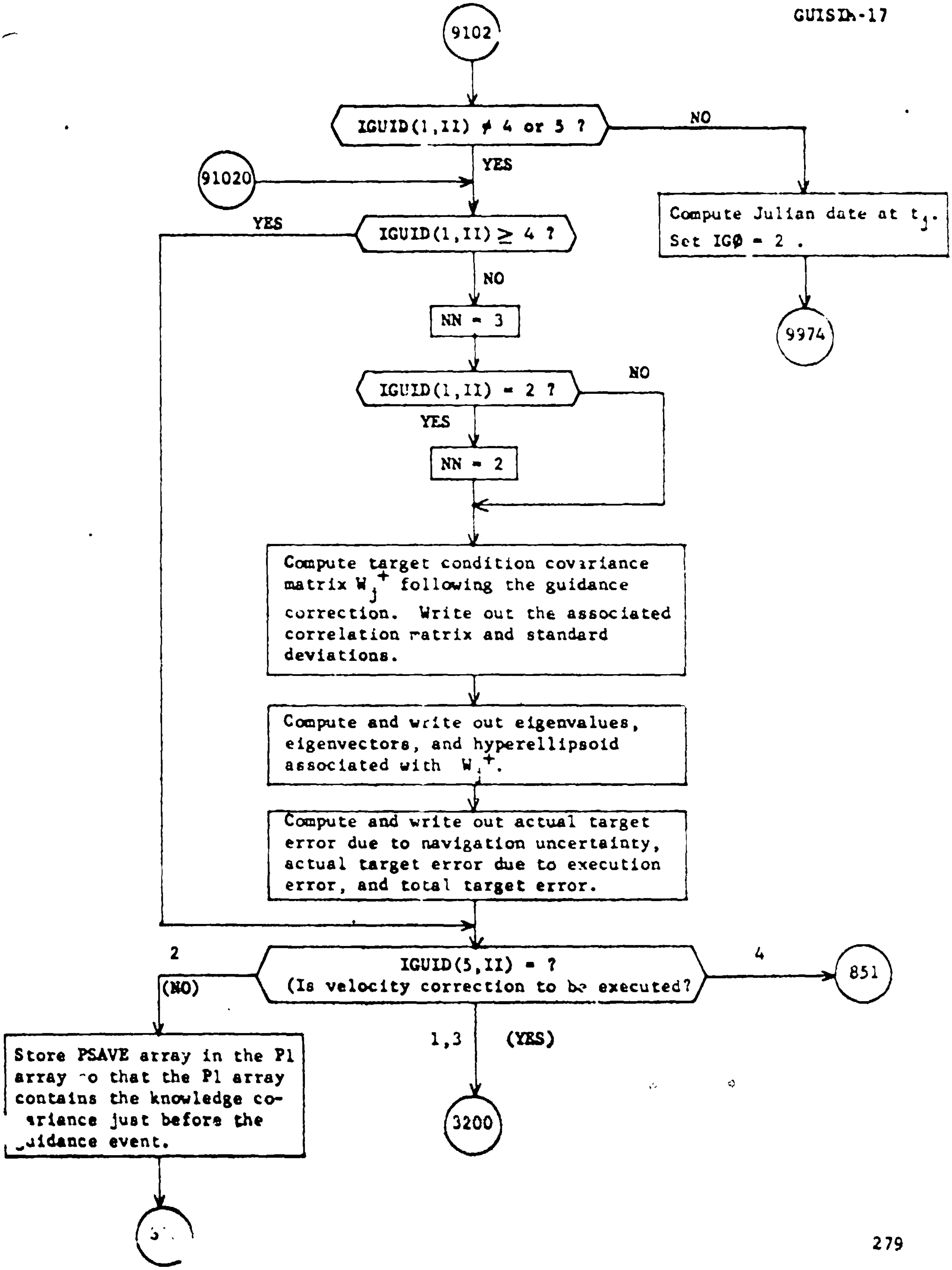

130
Compute and write out actual and estimated position/velocity deviations from the targeted nominal.
Compute and write out the commanded and perfect velocity corrections to null out errors from most recent target conditions. Compute and write out the error in the velocity correction due to navigation uncertainty.
861
NO
IGUID(3,II) = 0 ?
YES
Store $\Delta\hat{V}_{N_j}$ in DVN array for use in subroutine BIAIM.
Call BIAIM to perform biased aimpoint guidance event. Return aimpoint μ_j, bias velocity correction $\Delta\hat{V}_{UP_j}$, and execution error covariance matrix $\tilde{Q}_j$.
B
NO
t_j = injection time?
YES
Compute $\tilde{Q}_j$ using $\Delta\hat{V}_{N_j}$. Set IRET = 1.
Set $\tilde{Q}_j$ to zero.
9671
9760

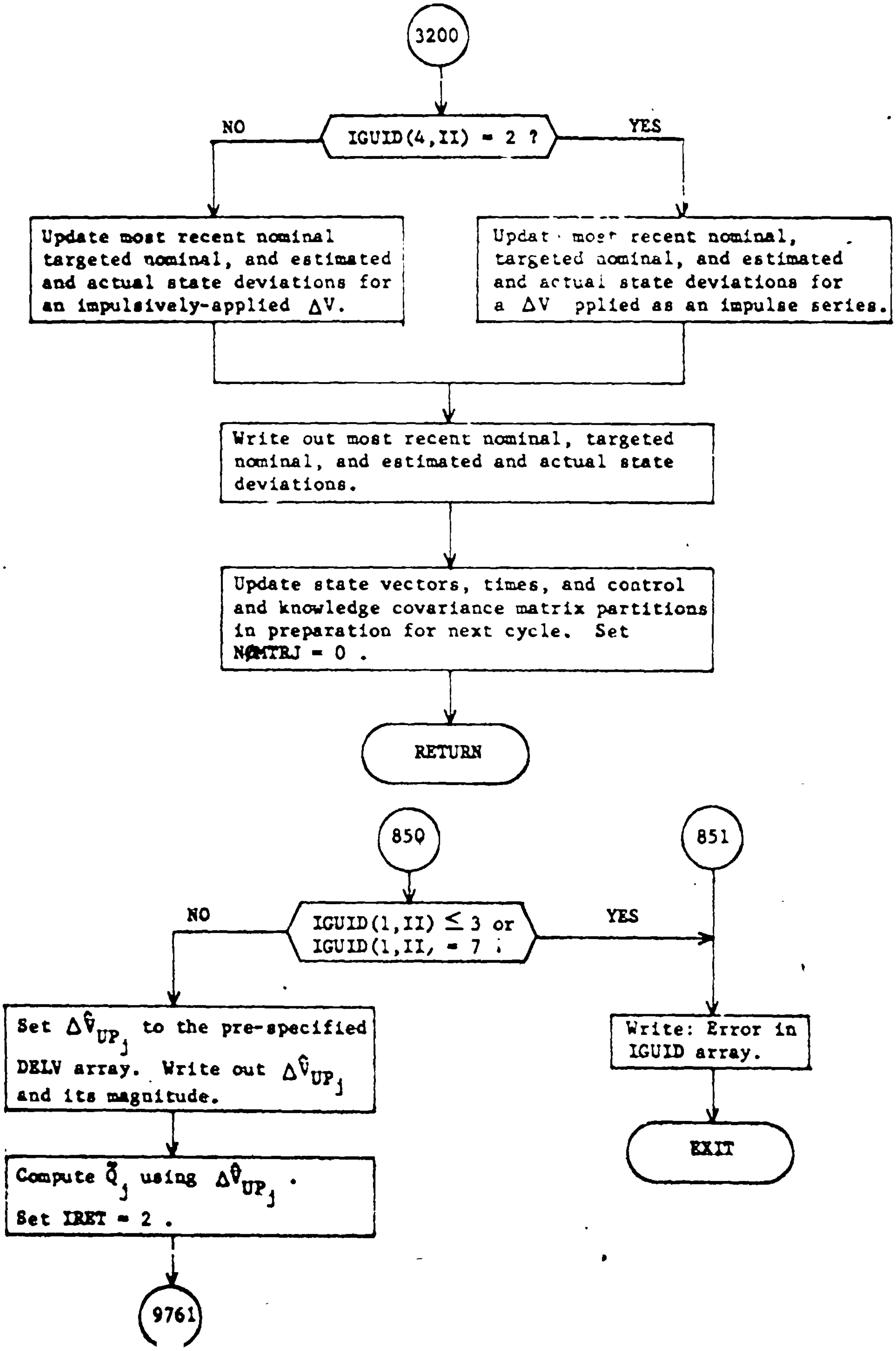

B
IGUID(3,II) = 1 ?
YES
900
NO
977
Set IX = 1 and JX = II.
Set XIN to estimated state at t_j.
Write: Commanded velocity correction will be recomputed using nonlinear guidance.
IGUID(1,II) = ?
1
FTA
2
2VBP
3
3VBP
Compute Julian date corresponding to t_{CA} and set IGØ = 1 .
9974
Compute position and velocity of target planet at the specified time.
IGØ = ?
2
99731
1
Define target variables XTAR, LKTAR, LNPAR, and TGT3 for nonlinear FTA guidance.
Define target variables XTAR, LKTAR, LNPAR, and TGT3 for nonlinear 2VBP guidance.
Define target variables XTAR, LKTAR, LNPAR, and TGT3 for nonlinear 3VBP guidance.
9980

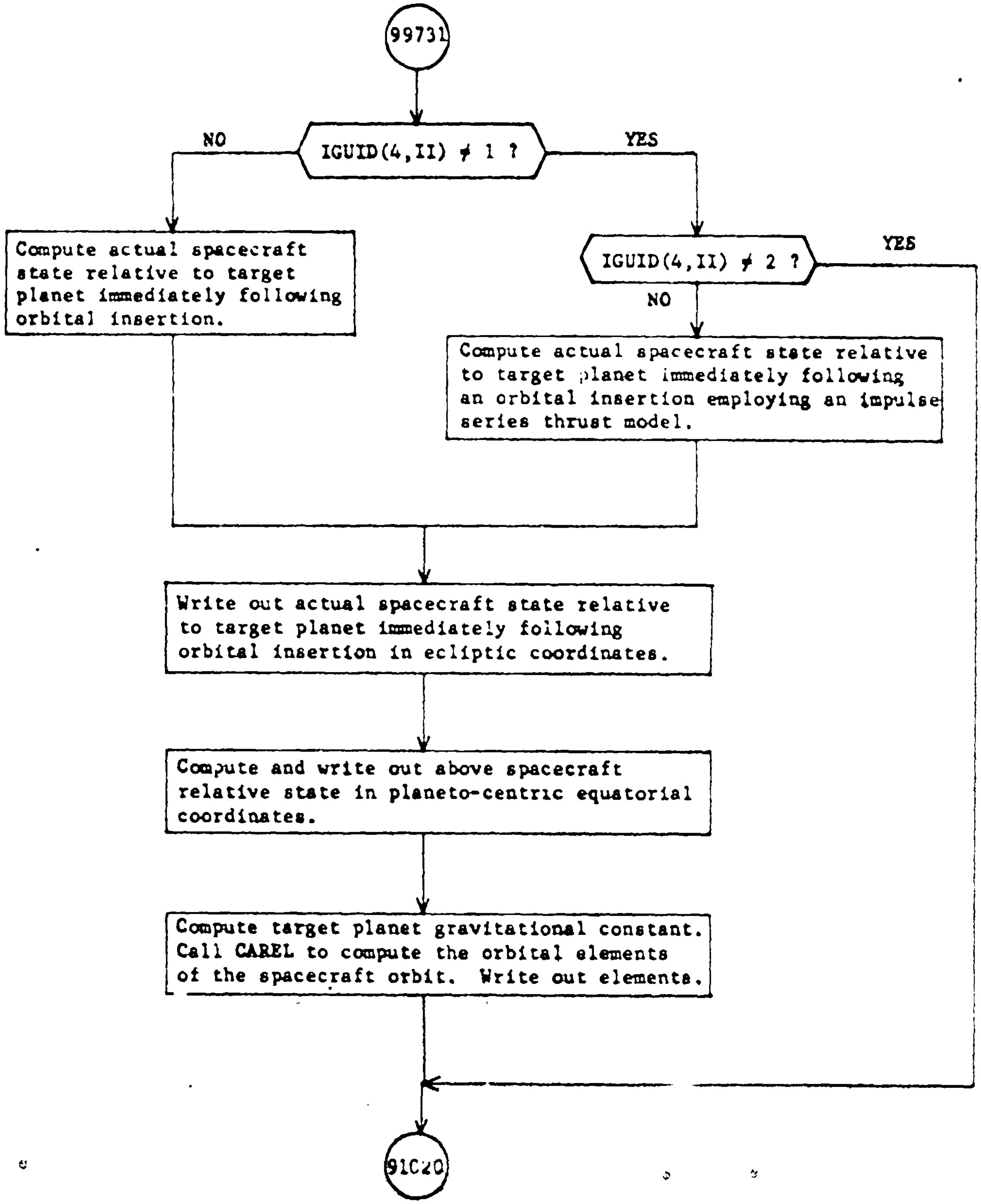

9980
Call NØNLIN to compute nonlinear commanded velocity correction. Return XDELV.
KWIT = 0 ?
NO
YES
Write: Nonlinear guidance failed. Linear guidance will be employed.
900
Set $\Delta\hat{V}_{UP_j}$ to XDELV. Compute magnitude. Write out $\Delta\hat{V}_{UP_j}$ and its magnitude.
t_j = injection time?
YES
NO
Set $\tilde{Q}_j = 0$.
Compute $\tilde{Q}_j$ using $\Delta\hat{V}_{UP_j}$. Set IRET = 2 .
9761
Write out correlation matrix and standard deviations associated with $\tilde{Q}_j$.
IRET = ?
2
3
1
320
900
Compute and write out eigenvalues, eigenvectors, and hyperellipsoid associated with $\tilde{Q}_j$.
9760
Set $\Delta\hat{V}_{UP_j}$ to $\Delta\hat{V}_{N_j}$.
IGUID(2,II) = 0 ?
YES
NO
900
977

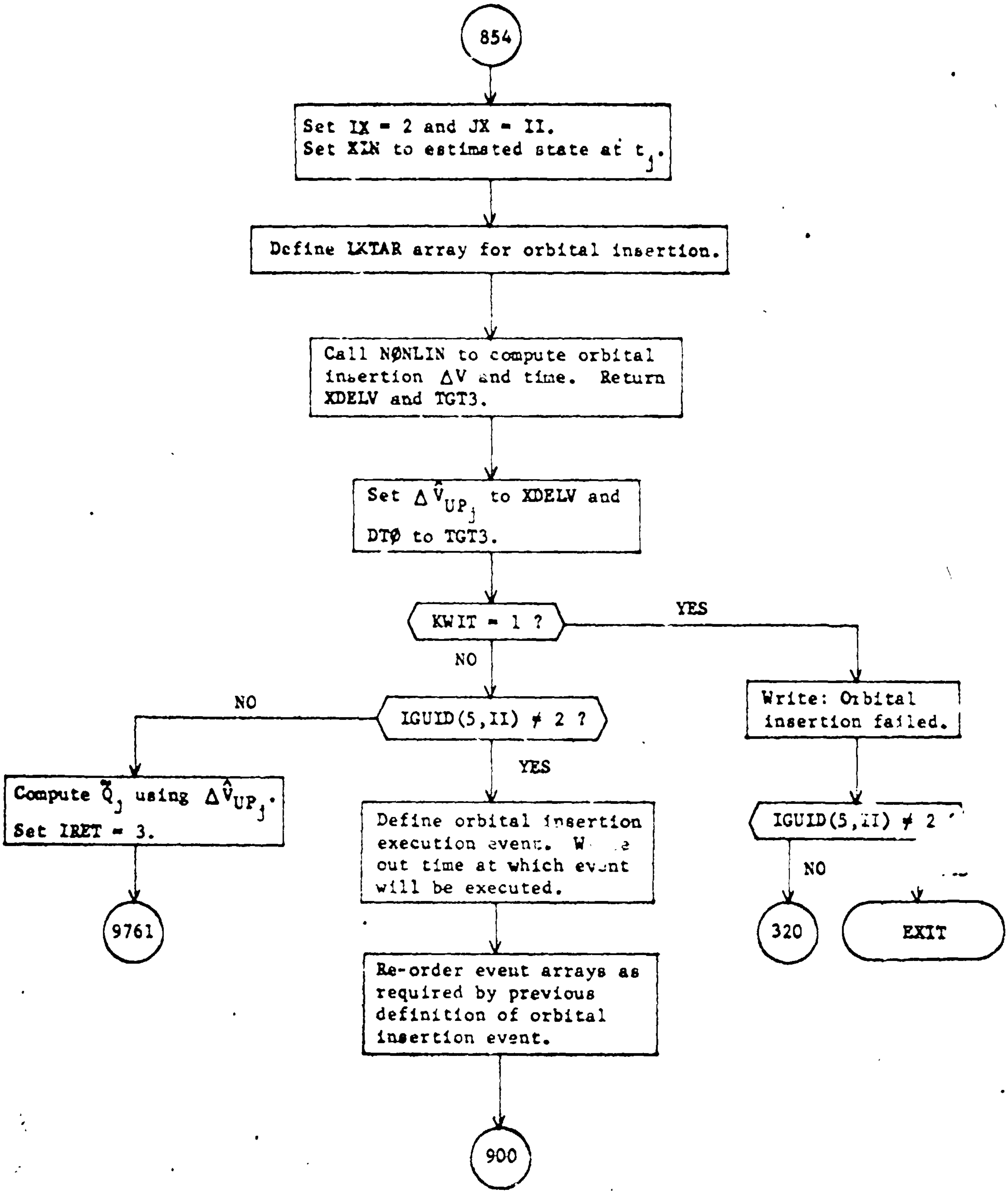

900
IGUID(5,II) = ?
(Is velocity correction to be treated at t_j?)
1,2,3
(YES)
4
(NO)
IGUID(1,II) = 4 or 5?
YES
NO
320
851
IGUID(4,II) = ?
(What kind of thrust model is to be employed?)
2
(impulse series)
904
3
(finite burn)
1
(single impulse)
Write: Finite burn not available.
EXIT
Store P1 array in PSAVE array. Update knowledge covariance by adding $\tilde{Q}_j$ to the velocity partition of the P1 array.
909
Call CØRREL to write out control and knowledge correlation matrix partitions and standard deviations just after the guidance correction.
ICØDE2 = 3
80

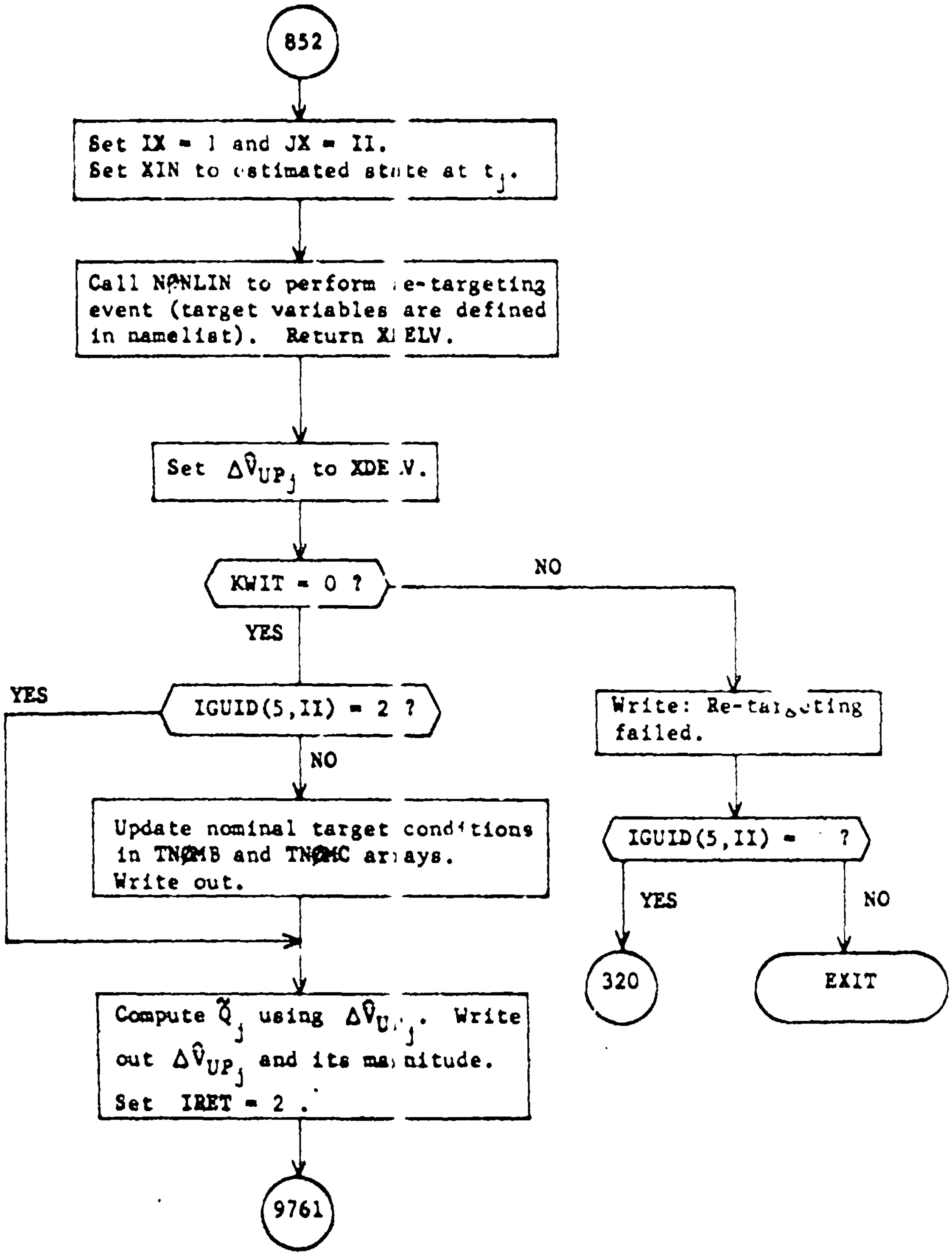

170
t_j = injection time?
YES
NO
$|\Delta\hat{V}_{UP_j}| \neq 0$?
NO
YES
Use $\Delta\hat{V}_{UF_j}$ to compute the actual execution error $\delta\Delta V_j$. Compute the actual velocity correction ΔV_j. Write out $\delta\Delta V_j$ and ΔV_j.
Set $\delta\Delta V_j$ and ΔV_j to zero.
IGUID(1,II) $\neq$ 4 or 5 ?
YES
NO
Compute and write out planeto-centric equatorial components of ΔV_j.
IGUID(4,II) $\neq$ 2 ?
YES
NO
XXIN to the actual sta.. .. t_j. Set DELPX to ΔV_j. Compute Julian date at t_j. Set INPX = 1 .
Call EXCUTS to compute the effective actual state. Return XXIN. Set XACT to XXIN.
9102

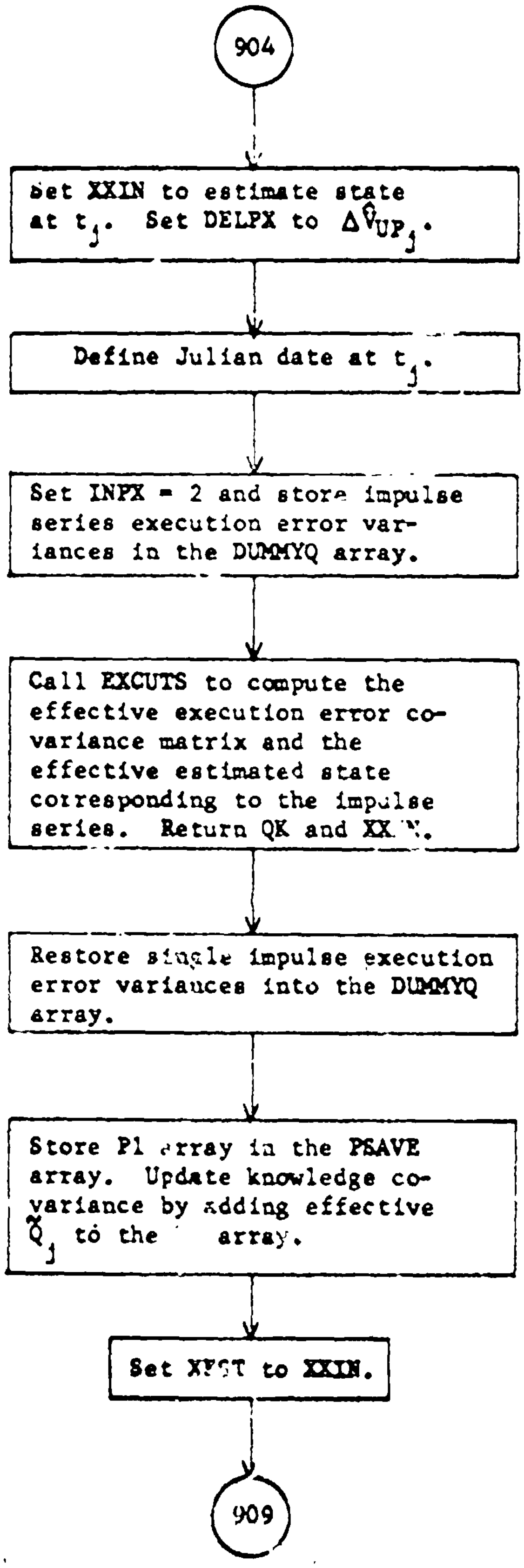

9102
IGUID(1,II) ≠ 4 or 5 ?
NO
YES
Compute Julian date at t_j.
Set IGØ = 2 .
9974
91020
YES
IGUID(1,II) ≥ 4 ?
NO
NN = 3
IGUID(1,II) = 2 ?
NO
YES
NN = 2
Compute target condition covariance matrix W_j^+ following the guidance correction. Write out the associated correlation matrix and standard deviations.
Compute and write out eigenvalues, eigenvectors, and hyperellipsoid associated with W_j^+.
Compute and write out actual target error due to navigation uncertainty, actual target error due to execution error, and total target error.
2
(NO)
IGUID(5,II) = ?
(Is velocity correction to be executed?
4
851
1,3
(YES)
Store PSAVE array in the P1 array so that the P1 array contains the knowledge covariance just before the guidance event.
3200


3200
IGUID(4,II) = 2 ?
NO
YES
Update most recent nominal targeted nominal, and estimated and actual state deviations for an impulsively-applied ΔV.
Updat· mos· recent nominal, targeted nominal, and estimated and actual state deviations for a ΔV pplied as an impulse series.
Write out most recent nominal, targeted nominal, and estimated and actual state deviations.
Update state vectors, times, and control and knowledge covariance matrix partitions in preparation for next cycle. Set NØMTRJ = 0 .
RETURN
850
851
IGUID(1,II) ≤ 3 or IGUID(1,II, = 7 ;
NO
YES
Set $\Delta\hat{V}_{UP_j}$ to the pre-specified DELV array. Write out $\Delta\hat{V}_{UP_j}$ and its magnitude.
Compute $\tilde{Q}_j$ using $\Delta\hat{V}_{UP_j}$. Set IRET = 2 .
9761
Write: Error in IGUID array.
EXIT


99731
IGUID(4,II) ≠ 1 ?
NO
YES
Compute actual spacecraft state relative to target planet immediately following orbital insertion.
IGUID(4,II) ≠ 2 ?
YES
NO
Compute actual spacecraft state relative to target planet immediately following an orbital insertion employing an impulse series thrust model.
Write out actual spacecraft state relative to target planet immediately following orbital insertion in ecliptic coordinates.
Compute and write out above spacecraft relative state in planeto-centric equatorial coordinates.
Compute target planet gravitational constant. Call CAREL to compute the orbital elements of the spacecraft orbit. Write out elements.
91020


854
Set IX = 2 and JX = II.
Set XIN to estimated state at t_j.
Define LXTAR array for orbital insertion.
Call NØNLIN to compute orbital insertion ΔV and time. Return XDELV and TGT3.
Set $\Delta\hat{V}_{UP_j}$ to XDELV and DTØ to TGT3.
KWIT = 1 ?
YES
NO
IGUID(5,II) ≠ 2 ?
NO
YES
Write: Orbital insertion failed.
Compute $\tilde{Q}_j$ using $\Delta\hat{V}_{UP_j}$.
Set IRET = 3.
9761
Define orbital insertion execution event.
out time at which event will be executed.
IGUID(5,II) ≠ 2
NO
320
EXIT
Re-order event arrays as required by previous definition of orbital insertion event.
900


852
Set IX = 1 and JX = II.
Set XIN to estimated state at t_j.
Call NØNLIN to perform re-targeting event (target variables are defined in namelist). Return XDELV.
Set $\Delta\hat{V}_{UP_j}$ to XDELV.
KWIT = 0 ?
NO
YES
IGUID(5,II) = 2 ?
YES
NO
Update nominal target conditions in TNØMB and TNØMC arrays. Write out.
Compute $\tilde{Q}_j$ using $\Delta\hat{V}_{UP_j}$. Write out $\Delta\hat{V}_{UP_j}$ and its magnitude. Set IRET = 2 .
9761
Write: Re-targeting failed.
IGUID(5,II) = ?
YES
NO
320
EXIT


904
Set XXIN to estimate state at t_j. Set DELPX to $\Delta\hat{V}_{UP_j}$.
Define Julian date at t_j.
Set INPX = 2 and store impulse series execution error variances in the DUMMYQ array.
Call EXCUTS to compute the effective execution error covariance matrix and the effective estimated state corresponding to the impulse series. Return QK and XX.N.
Restore single impulse execution error variauces into the DUMMYQ array.
Store P1 array in the PSAVE array. Update knowledge covariance by adding effective $\tilde{Q}_j$ to the array.
Set XEST to XXIN.
909

SUBROUTINE HELIO

PURPOSE: TO COMPUTE THE ZERO ITERATE INJECTION STATE FOR INTERPLANETARY TARGETING

CALLING SEQUENCE: CALL HELIO

SUBROUTINES SUPPORTED: ZERIT

SUBROUTINES REQUIRED: LAUNCH FLITE ELCAR EPHEM ORB PECEQ TIME

LOCAL SYMBOLS:

AHEL	SEMI-MAJOR AXIS OF THE HELIOCENTRIC CONIC
ARGP	ARGUMENT OF PERIAPSIS OF THE HELIOCENTRIC CONIC IN RADIANS
ASCND	LONGITUDE OF THE ASCENDING NODE OF THE HELIOCENTRIC CONIC IN RADIANS
ATP	SEMI-MAJOR AXIS OF TARGET PLANETOCENTRIC CONIC
AZF	AZIMUTH AT DF ON THE HELIOCENTRIC CONIC IN DEGREES
AZI	AZIMUTH AT DI ON THE HELIOCENTRIC CONIC IN DEGREES
B2	SQUARE OF THE B VECTOR MAGNITUDE OF THE TARGET PLANETOCENTRIC CONIC
CBDR	DESIRED B.R MAGNITUDE AT DF OF THE TARGET PLANETOCENTRIC CONIC
CBDT	DESIRED B.T MAGNITUDE AT DF OF THE TARGET PLANETOCENTRIC CENTER
COSASN	CONSINE OF ASCND
COSB	INTERMEDIATE VARIABLE FOR AZI,AZF EQUATION
COSFS	COSINE OF FS
COSF	COSINE OF TAI
COSPSI	COSINE OF PSI
COSTHE	COSINE OF THETAI
CRCA	DESIRED RCA MAGNITUDE AT DF OF THE TARGET PLANETOCENTRIC CONIC

DELT	TIME OF FLIGHT (SECS) OF HELICOENTRIC CONIC
DF	FINAL JULIAN DATE OF HELIOCENTRIC CONIC
DI	INITIAL JULIAN DATE OF HELIOCENTRIC CONIC
DSICA	DELTA TIME (DAYS) FROM SPHERE-OF-INFLUENCE TO CLOSEST APPROACH OF THE TARGET PLANET OCENTRIC CONIC
EHEL	ECCENTRICITY OF THE HELIOCENTRIC CONIC
EQEC	TRANSFORMATION MATRIX FROM ECLIPTIC TO TARGET PLANET EQUATORIAL AT DF
ETP	ECCENTRICITY OF THE TARGET PLANETOCENTRIC CONIC
FF	INTERMEDIATE VARIABLE FOR COMPUTATION OF DSICA
FS	TRUE ANOMALY OF THE TARGET PLANETOCENTRIC CONIC
IDAT	CALENDER DATE CORRESPONDING TO DF
IDUM	DUMMY ARGUMENT FOR CALL TO SUBROUTINE FLITE
ITIM	INDICATES COMPUTATION OF HELIOCENTRIC STATES =0, COMPUTE INITIAL AND FINAL STATES =1, COMPUTE FINAL STATE ONLY
I	INDEX
J	INDEX
LDAT	CALENDER DATE CORRESPONDING TO DI
OAPO	APOAPSIS RADIUS OF THE HELIOCENTRIC CONIC
OASN	ASCND CONVERTED TO DEGREES
OECC	OUTPUT ECCENTRICITY OF THE HELIOCENTRIC CONIC
OGAF	FLIGHT PATH ANGLE AT DF OF THE HELIO-CENTRIC CONIC
OGAI	FLIGHT PATH ANGLE AT DI OF THE HELIO-CENTRIC CONIC

OHCA	CENTRAL ANGLE OF THE HELIOCENTRIC CONIC
OINC	INCLINATION OF THE HELIOCENTRIC CONIC
OLAF	LATITUDE AT DF OF HELIOCENTRIC CONIC
OLAI	LATITUDE AT DI OF HELIOCENTRIC CONIC
OLOF	LONGITUDE AT DF OF HELIOCENTRIC CONIC
OLOI	LONGITUDE AT DI OF HELIOCENTRIC CONIC
OPER	ARGUMENT OF PERIAPSIS OF THE HELIOCENTRIC CONIC IN DEGREES
ORCA	PERIAPSIS RADIUS OF HELIOCENTRIC CONIC
ORF	MAGNITUDE OF HELIOCENTRIC POSITION AT DF IN OUTPUT UNITS
ORI	MAGNITUDE OF HELIOCENTRIC POSITION AT DI IN OUTPUT UNITS
OSMA	SEMI-MAJOR AXIS OF THE HELIOCENTRIC CONIC IN OUTPUT UNITS
OTAF	TRUE ANOMALY AT DF OF HELIOCENTRIC CONIC IN DEGREES
OTAI	TRUE ANOMALY AT DI OF HELIOCENTRIC CONIC IN DEGREES
OVF	MAGNITUDE OF HELIOCENTRIC VELOCITY AT DF IN KILOMETERS
OVI	MAGNITUDE OF HELIOCENTRIC VELOCITY AT DI IN KILOMETERS
OVPF	VELOCITY OF TARGET PLANET AT DF
OVPI	VELOCITY OF TARGET PLANET AT DI
PHEL	SEMI-LAUS RECTUM OF HELIOCENTRIC CONIC
PLINC	INCLINATION (IN RADIANS) OF HELIOCENTRIC CONIC
PSI	CENTRAL ANGLE (IN RADIANS) OF HELIOCENTRIC CONIC
PTP	SEMI-LATUS RECTUM OF TARGET PLANETOCENTRIC CONIC AT DF, USED TO CALCULATE OSICA

RF	MAGNITUDE OF TARGET PLANET POSITION AT DF
RI	MAGNITUDE OF TARGET PLANET POSITION AT DI
RTM	MAGNITUDE OF RT VECTOR
RT	HELIOCENTRIC POSITION VECTOR OF THE FINAL CONIC CORRESPONDING TO OTAF
RZM	MAGNITUDE OF THE RZ VECTOR
RZ	HELIOCENTRIC POSITION VECTOR OF THE FINAL CONIC CORRESPONDING TO OTAI
SGN	INTERNAL SIGN VARIABLE USED TO DEFINE THE TRANSFER PLANE ORIENTATION
SINASN	SINE OF ASCND
SINF	SIN OF TAI
SINHF	HYPERBOLIC SINE OF THE AUXILIARY VARIABLE F USED TO CALCULATE DSICA
SINPSI	SINE OF PSI
SI	SECONDS IN CALENDER DATE IDAT
SL	SECONDS IN CALENDER DATE LDAT
SUNMU	GRAVITATIONAL CONSTANT OF SUN IN KM**3/SEC**2
TAF	OTAF IN RADIANS
TAI	OTAI IN RADIANS
TANF	TANGENT OF THE AUXILIARY VARIABLE F USED TO CALCULATE DSICA
TERM	INTERMEDIATE VARIABLE USED TO CALCULATE CRCA
TEST	INTERMEDIATE VARIABLE USED TO CALCULATE AZIMUTHS AND PATH ANGLES
TFP	DUMMY VARIABLE USED TO CALL ELCAR
THETAI	INTERMEDIATE ANGLE USED TO DEFINE ARGP
TSPH	SPHERE-OF-INFLUENCE OF TARGET PLANET IN KILOMETERS

VF	VELOCITY OF THE TARGET PLANET AT DF
VHAT	INTERMEDIATE VECTOR USED TO DEFINE AZI, AZF
VHP	HYPERBOLIC EXCESS VELOCITY OF THE TARGET PLANETOCENTRIC CONIC AT DF
VHPM	MAGNITUDE OF THE VHP VECTOR USED TO CALCULATE DSICA
VI	VELOCITY OF LAUNCH PLANET AT DI
VMAG	INTERMEDIATE VARIABLE USED TO DEFINE AZI, AZF
VTM	MAGNITUDE OF VT VECTOR
VT	HELIOCENTRIC VELOCITY VECTOR OF THE FINAL CONIC CORRESPONDING TO OTAF
VZM	MAGNITUDE OF THE VZ VECTOR
VZ	HELIOCENTRIC VELOCITY VECTOR OF THE FINAL CONIC CORRESPONDING TO OTAI
WHAT	UNIT VECTOR NORMAL TO THE TRANSFER PLANE
WMAG	MAGNITUDE OF THE NON-UNITIZED WHAT VECTOR
XF	POSITION OF THE TARGET PLANET AT DF
XI	POSITION OF THE LAUNCH PLANET AT DI

COMMON COMPUTED/USED:	TMU	VHPM			
COMMON COMPUTED:	DPA	NO	RAP	RIN	TIN
COMMON USED:	ALNGTH	DG	DT	IZERO	KTAR
	KUR	NLP	NTP	ONE	PI
	PMASS	RAD	SPHERE	TAR	TM
	TWO	XP	ZDAT	ZERO	

HELIO Analysis

HELIO computes the zero iterate initial state for interplanetary trajectories. The initial and final states are determined either by an arbitrary position vector or by the location of a specified planet at a . escribed time according to

IZERO = 1 planet to planet
2 planet to arbitrary final point
3 arbitrary initial point to planet
4 arbitrary initial point to final point

The final time used in locating a planet must correspond to the closest approach (CA) to the planet. Therefore if the target time is read in as a sphere of influence (SOI) time, a modification is required. The heliocentric conic is computed (as described below) using the t_{SI} time to determine the final position. The approach asymptote $\vec{V}_{HP}$ corresponding to that trajectory is used with the desired r_{CA} to compute the time from SOI to CA. If r_{CA} is not a target variable then the target values of B·T and B·R are used to estimate the r_{CA}

$$r_{CA} = -\frac{\mu}{V_{HP}} + \frac{1}{2}\sqrt{\left(\frac{2\mu}{V_{HP}^2}\right)^2 + 4B^2} \qquad (1)$$

Then the approximate approach hyperbola is given by

$$a_h = \frac{\mu r_{SI}}{2\mu - V_{HP}^2 r_{SI}}$$

$$e_h = 1 - \frac{r_{CA}}{a} \qquad (2)$$

$$p_h = a_h(1 - e_h^2)$$

and the hyperbolic time to go from SOI to CA is given by

$$\Delta t_{SICA} = \frac{\mu}{V_{HP}^3}\ (e \sinh F - F) \qquad (3)$$

where

$$\tanh \frac{F}{2} = \sqrt{\frac{e_h - 1}{e_h + 1}} \tanh \frac{f}{2}$$

$$\cos f = \frac{1}{e} \left(\frac{P_h}{r_{SI}} - 1 \right) \tag{4}$$

The final time is then given by $t_f = t_{SI} + \Delta t_{SICA}$.

The initial and final positions $\vec{r}_i$ and $\vec{r}_f$ of the heliocentric conic are either input or computed from the positions of planets determined by ORB and EPHEM. The unit normal to the heliocentric orbit plane is

$$\hat{W} = \frac{\vec{r}_i \times \vec{r}_f}{\left| \vec{r}_i \times \vec{r}_f \right|} \tag{5}$$

The inclination to that plane is

$$\cos i = \hat{W}_z \tag{6}$$

The ascending node of the plane is given by

$$\tan \Omega = \frac{\hat{W}_x}{-\hat{W}_y} \tag{7}$$

The central angle of transfer is defined by

$$\cos \Psi = \frac{\vec{r}_i \cdot \vec{r}_f}{r_i r_f} \tag{8}$$

The semi-major axis a and eccentricity e of the heliocentric conic are computed from Lambert's theorem in subroutine FLITE. The true anomaly f_i at the initial and final points are computed from

$$p = a (1 - e^2)$$

$$\cos f_i = \frac{p - r_i}{e\, r_i} \qquad \sin f_i = \frac{\cos f_i \cos \Psi - \frac{p - r_f}{e\, r_f}}{\sin \Psi} \tag{8}$$

$$f_f = f_i + \Psi$$

Finally, the argument of periapsis ω is computed from

$$\cos(\omega + f_i) = \frac{\vec{r}_i \cdot \hat{U}}{r_i} \tag{10}$$

where $\hat{U} = (\cos \Omega \ , \ \sin \Omega \ , \ 0)$.

Therefore the initial or final states ($\vec{r}_i$, $\vec{v}_i$) or ($\vec{r}_f$, $\vec{v}_f$) may now be computed by ELCAR. Let ($\vec{r}$, $\vec{v}$) denote either state and let ($\vec{r}$, $\vec{v}_p$) denote the state of the relevant planet. The departure (or approach) asymptote is then given by

$$\vec{v}_{HP} = \vec{v}_f - \vec{v}_p \qquad \vec{v}_{HE} = \vec{v}_i - \vec{v}_p \tag{11}$$

The latitude and longitude of the position vector are

$$\sin \phi = \frac{\vec{z}}{r} \qquad \tan \theta = \frac{r_y}{r_x} \tag{12}$$

The path angle Γ may be computed from

$$\cos \Gamma = \frac{\sqrt{\mu p}}{r v} \tag{13}$$

The azimuth of the relevant state is

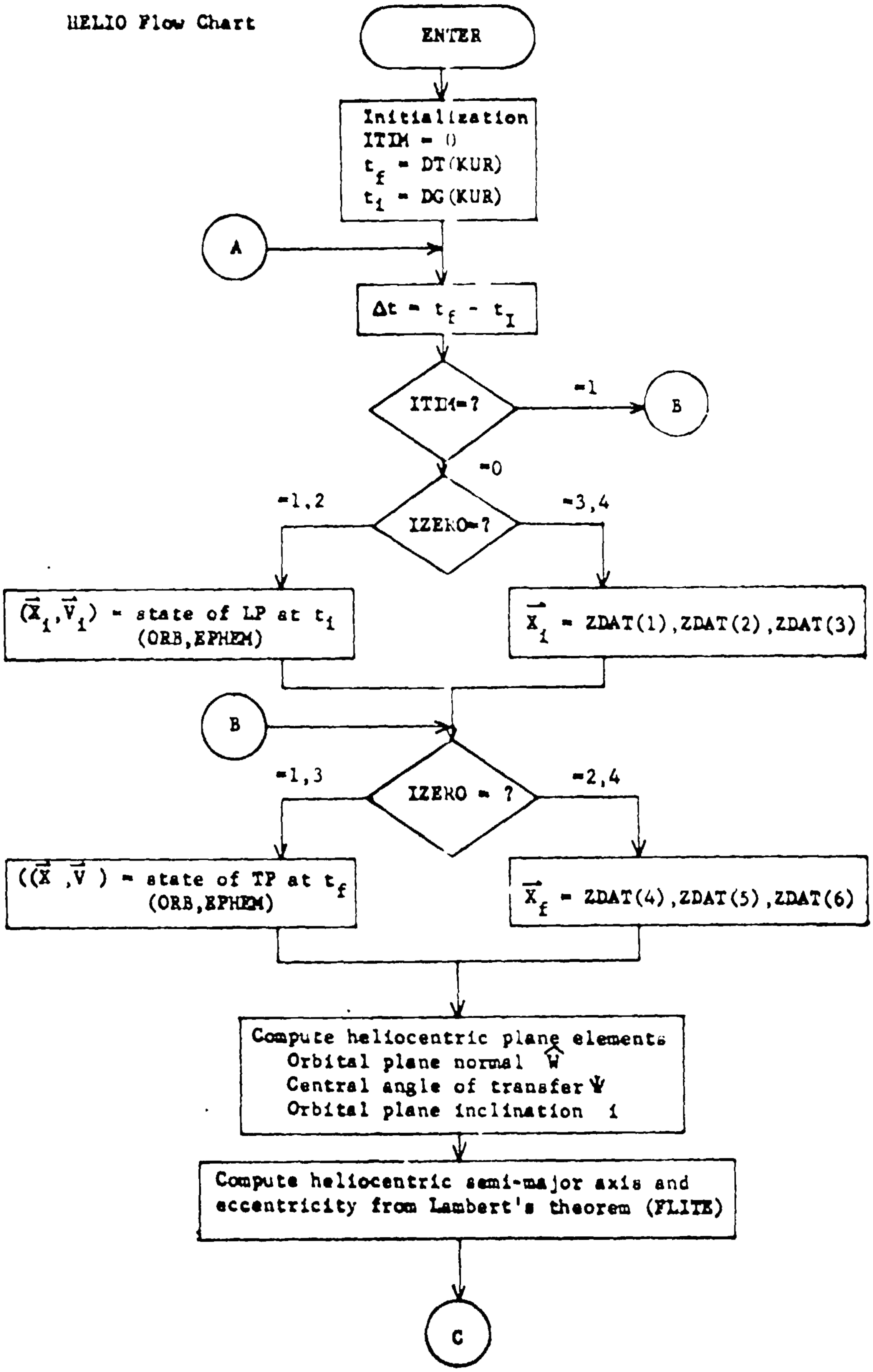
$$\sin \Sigma = \frac{(\vec{r} \times \vec{v}) \cdot \hat{U}}{|\vec{r} \times \vec{v}|} \tag{14}$$

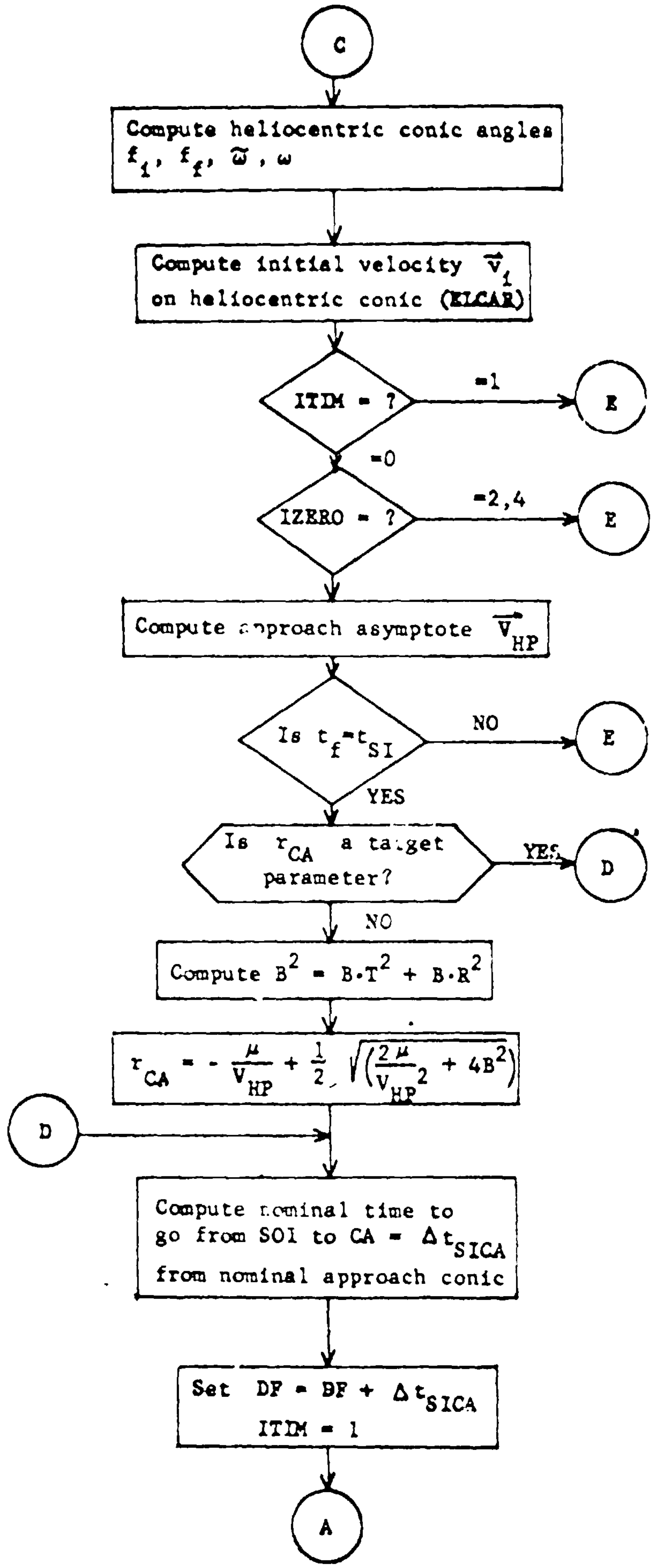
$$\cos \Sigma = \frac{\vec{V} \cdot \hat{U}}{V \cos \Gamma} \tag{15}$$

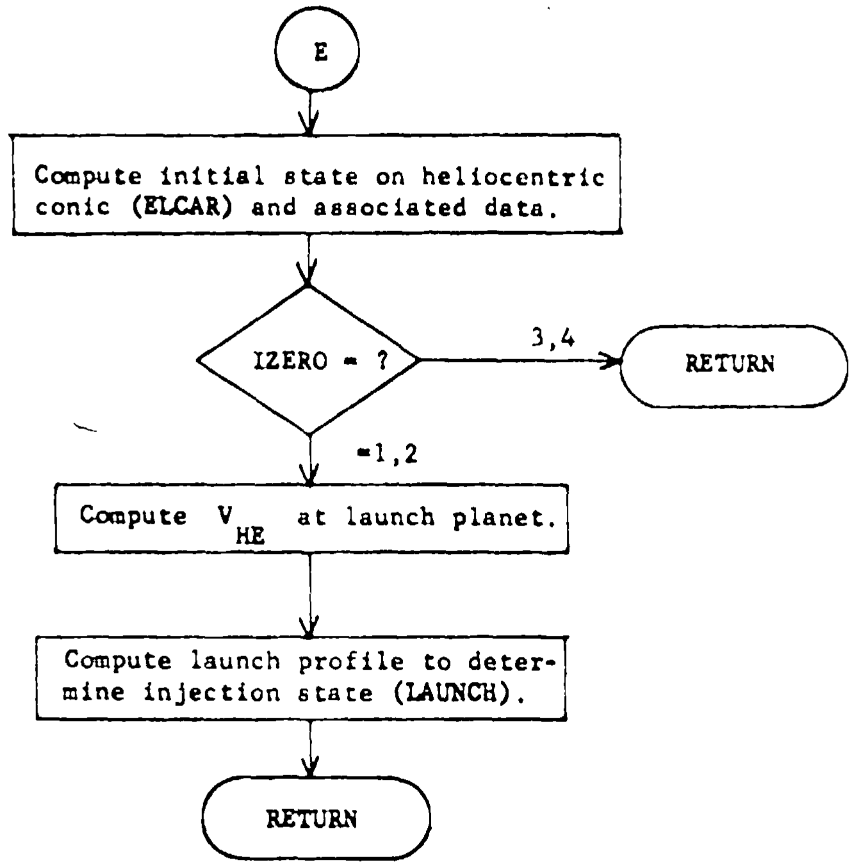
If the initial state is referenced to a planet, subroutine LAUNCH is called to convert the departure asymptote and launch profile into an injection radius, velocity, and time. Otherwise the initial state is returned as the initial state on the heliocentric conic.

Reference: Space Research Conic Program, Phase III, May 1, 1969, Jet Propulsion Laboratory, Pasadena, California.

HELIO Flow Chart




C
Compute heliocentric conic angles f_i, f_f, $\tilde{\omega}$, ω
Compute initial velocity $\vec{V}_i$ on heliocentric conic (ELCAR)
ITIM = ?
=1
E
=0
IZERO = ?
=2,4
E
Compute approach asymptote $\vec{V}_{HP}$
Is $t_f = t_{SI}$
NO
E
YES
Is r_{CA} a target parameter?
YES
D
NO
Compute $B^2 = B{\cdot}T^2 + B{\cdot}R^2$
$r_{CA} = -\frac{\mu}{V_{HP}} + \frac{1}{2}\sqrt{\left(\frac{2\mu}{V_{HP}^2}\right)^2 + 4B^2}$
D
Compute nominal time to go from SOI to CA = Δt_{SICA} from nominal approach conic
Set DF = BF + Δt_{SICA}
ITIM = 1
A


E
Compute initial state on heliocentric conic (ELCAR) and associated data.
IZERO = ?
3,4
RETURN
=1,2
Compute V_{HE} at launch planet.
Compute launch profile to determine injection state (LAUNCH).
RETURN

SUBROUTINE HYELS

PURPOSE: TO COMPUTE AND PRINT THE TWO-DIMENSIONAL OR THREE-DIMENSIONAL HYPERELLIPSOID OF A SPECIFIED MATRIX.

CALLING SEQUENCE: CALL HYELS(KS,P,N)

ARGUMENT:	KS	I	SIGMA LEVEL OF THE HYPERELLIPSOID
	P	I	MATRIX FOR WHICH THE HYPERELLIPSOID IS TO BE COMPUTED
	N	I	DIMENSION LIMITS OF THE SQUARE MATRIX P

SUBROUTINES SUPPORTED: EIGHY GUISIM GUISS SETEVN GUIDM GUID PRED

SUBROUTINES REQUIRED: MATIN

LOCAL SYMBOLS:	K2	SQUARE OF SIGMA LEVEL
	PI	INVERSE OF MATRIX P
	P12	TWICE THE VALUE OF (1,2) ELEMENT OF PI
	P13	TWICE THE VALUE OF (1,3) ELEMENT OF PI
	P23	TWICE THE VALUE OF (2,3) ELEMENT OF PI
	V	TEMPORARY STORAGE VECTOR FOR ARRAY P

COMMON USED: TWO

HYELS Analysis

Subroutine HYELS computes and writes out hyperellipsoids associated with a 2 or 3 dimensional covariance matrix P.

If P is assumed to be the covariance matrix of an n-dimensional random variable $\hat{x}$ having a gaussian distribution with mean zero, then the probability density function is given by

$$p = \frac{1}{(2\pi)^{n/2} |P|^{1/2}} \exp \left[- \frac{1}{2} \hat{x}^T P^{-1} \hat{x} \right]$$

Re-writing this equation as

$$\hat{x}^T P^{-1} \hat{x} = 2 \ln \left[\frac{1}{(2\pi)^{n/2} p \; |P|^{1/2}} \right] = k^2$$

shows that the surface of constant probability density p is an m-dimensional ellipsoid, where m is the rank of P. The constant k can be shown to correspond to the sigma level of the ellipsoid.

For n = 3, the above equation has form

$$ax^2 + by^2 + cz^2 + d\,xy + e\,xz + f\,yz = k^2$$

where $a = a_{11}$ $\quad d = 2a_{12}$

$b = a_{22}$ $\quad e = 2a_{13}$

$c = a_{33}$ $\quad f = 2a_{23}$

and the a_{ij} are the elements of P^{-1}.

Subroutine HYELS uses this equation to compute a 3-dimensional hyperellipsoid, and sets the appropriate constants to zero to compute a 2-dimensional hyperellipsoid.

Reference: H. Sorenson. "Kalman Filtering", Advances in Control Systems, Vol. 3, C. T. Leades (Ed.). New York: Academic Press, 1966, p. 219.

SUBROUTINE IMPACT

PURPOSE: TO COMPUTE THE ACTUAL IMPACT PLANE PARAMETERS BDT AND BDR CORRESPONDING TO ANY POINT ON AN INCOMING HYPERBOLA. IT HAS THE OPTION TO CONVERT TARGET VALUES OF INCLINATION XIN AND RADIUS OF CLOSEST RCA INTO EQUIVALENT TARGET VALUES OF DBT AND DBR.

CALLING SEQUENCE: CALL IMPACT(R,V,GMX,T,BDT,BDR,XIN,RCA,DBT,DBR, TCA,KOPT)

ARGUMENTS

R(3)	I	POSITION VECTOR TO CENTRAL BODY AT EPOCH
V(3)	I	VELOCITY VECTOR TO CENTRAL BODY AT EPOCH
GMX	I	GRAVITATIONAL CONSTANT OF CENTRAL BODY
T(3,3)	I	TRANSFORMATION MATRIX FROM REFERENCE TO INCLINATION SYSTEM
BDT	O	VALUE OF ACTUAL B.T EVALUATED AT EPOCH
BDR	O	VALUE OF ACTUAL B.R EVALUATED AT EPOCH
XIN	I	DESIRED INCLINATION (DEG) (OPTIONAL)
RCA	I	DESIRED RADIUS OF CLOSEST APPROACH (OPTION)
DBT	O	TARGET VALUE OF B.T BASED ON XIN, RCA
DBR	O	TARGET VALUE OF B.R BASED ON XIN, RCA
TCA	O	TIME FROM PERIAPSIS ON CONIC
KOPT	I	TARGET VALUE COMPUTATION FLAG =0 DO NOT COMPUTE TARGET VALUES =1 COMPUTE TARGET VALUES OF B.T, B.R (MUST READ IN OPTIONAL INPUT)

SUBROUTINES SUPPORTED: TAROPT LUNCON LUNTAR MULTAR VMP

SUBROUTINES REQUIRED: NONE

LOCAL SYMBOLS:

AB	INTERMEDIATE VARIABLE FOR CALCULATION OF RV,SV,TV SYSTEM
AIN	TARGET INCLINATION IN RADIANS. AFTER NORMALIZATION
ANG	OUTPUT VARIABLE WHEN DECLINATION CONSTRAINT IS VIOLATED

AUXF	ECCENTRIC ANOMALY (HYPERBOLIC CASE)
A	SEMI-MAJOR AXIS OF R-V CONIC
BMAG	MAGNITUDE OF DESIRED B VECTOR
BV	ACTUAL/DESIRED B VECTOR
B	MAGNITUDE OF ACTUAL B VECTOR
CDECL	COSINE OF DECL
CDELW	COSINE OF DELW
CTA	COSINE OF TA
CW	COSINE OF W
C1	MAGNITUDE OF VECTOR NORMAL TO ORBITAL PLANE IN INERTIAL SYSTEM
DB	DESIRED MAGNITUDE OF DESIRED B VECTOR
DECL	DECLINATION OF APPROACH ASYMPTOTE IN INCLINATION SYSTEM
DELW	LONGITUDE OF ASCENDING NODE IN INCLINATION SYSTEM
E	ECCENTRICITY OF THE R-V CONIC
II	INCLINATION SIGN INDICATOR. =1, INCLINATION IS POSITIVE =-1, INCLINATION IS NEGATIVE
IM	INDICATOR FOR DIRECTION OF MOTION OF THE TRAJECTORY =1, MOTION IS POSIGRADE =-1,MOTION IS RETROGRADE
PI	MATHEMATICAL CONSTANT 3.141592653589793
PV	INTERMEDIATE VECTOR USED TO CALCULATE DESIRED B VECTOR
P	SEMI-LATUS RECTUM
QV	INTERMEDIATE VECTOR USED TO CALCULATE ACTUAL B VECTOR
RAD	DEGREES TO RADIANS CONVERSION CONSTANT

RD	TIME DERIVATIVE OF RM
RM	MAGNITUDE OF THE POSITION VECTOR R
RRD	DOT PRODUCT OF R AND V VECTORS
RV	VECTOR USED TO CALCULATE ACTUAL AND DESIRED B DOT R
SDECL	SINE OF DECL
SDELW	SINE OF DELW
SINHF	HYPERBOLIC SINE OF AUXF
STA	SINE OF TA
SV	VECTOR USED TO CONSTRUCT RV, TV VECTORS. PARALLEL TO THE APPROACH ASYMPTOTE
SW	SINE OF W
SX	VARIABLE USED TO DETERMINE SIGNS OF DBT, DBR
TANG	INTERMEDIATE VARIABLE FOR CALCULATION OF AUXF
TA	TRUE ANOMALY FOR CALCULATION OF AUXF
THS	INTERMEDIATE ANGLE FOR CALCULATION OF W
TV	VECTOR USED TO CALCULATE AC.UAL AND DESIRED B DOT T
VINH	VELOCITY AT INFINITY
VX	MAGNITUDE OF THE VELOCITY VECTOR V
WMAG	MAGNITUDE OF VECTOR NORMAL TO ORBITAL PLANE IN INCLINATION SYSTEM
WV	VECTOR NORMAL TO ORBITAL PLANE IN INCLINATION AND INERTIAL SYSTEMS
W	ARGUMENT OF PERIAPSIS
Z	APPROACH ASYMPTOTE IN INCLINATION SYSTEM

COMMON USED. NINETY ONE TWO ZERO

IMPACT Analysis

The impact parameters $B \cdot T$ and $B \cdot R$ form a convenient set of variables for the description of the approach geometry for lunar and interplanetary missions. Let a reference cartesian coordinate system XYZ (ecliptic in STEAP) be established at the center of the target body. Let $\vec{V}_\infty$ denote the hyperbolic excess velocity of the spacecraft in the XYZ system. An auxiliary coordinate system R-S-T may be constructed relative to the $\vec{V}_\infty$ by the definitions

$$\hat{S} = \vec{V}_\infty / V_\infty \qquad \hat{T} = \frac{\hat{S} \times \hat{K}}{|\hat{S} \times \hat{K}|} \qquad \hat{R} = \hat{S} \times \hat{T} \tag{1}$$

Therefore $\hat{S}$ is in the direction of the approach asymptote, $\hat{T}$ lies along the intersection of the impact plane (the plane normal to $\hat{S}$ and passing through the center of the planet) and the reference plane (XY-plane), and R completes the right hand system. The $\vec{B}$ vector lies in the impact plane and is directed to the incoming asymptote. Then $B \cdot T$ and $B \cdot R$ have the usual vector definitions.

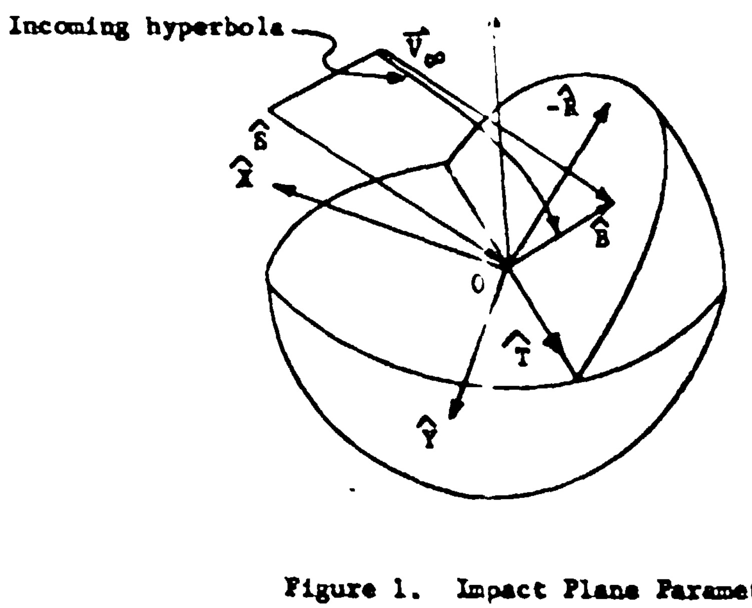


Figure 1. Impact Plane Parameters

In the optional part of the subroutine, the target impact parameter $\vec{B}^*$ associated with $\hat{S}$ and a target inclination i (relative to target planet equator) and radius of closest approach r_{CA} is computed. However given an approach asymptote $\hat{S}$ there are generally four trajectories with the same values of i and r_{CA}. Two of these trajectories are retrograde and

two are posigrade. For each type of motion there are two distinct planes that have the same inclination and include the $\hat{S}$ vector. These are distinguished by the direction of motion when the approach asymptote is crossed, i.e., whether the motion is from north to south (northern approach) or from south to north (southern approach). Let $0 \leq \alpha \leq 90^\circ$. Then setting the target inclinations to the following values determines the trajectory which will be specified:

i	Trajectory
α	posigrade with northern approach
$-\alpha$	posigrade with southern approach
$180+\alpha$	retrograde with northern approach
$180-\alpha$	retrograde with southern approach

The possible trajectories are illustrated in Figure 2.

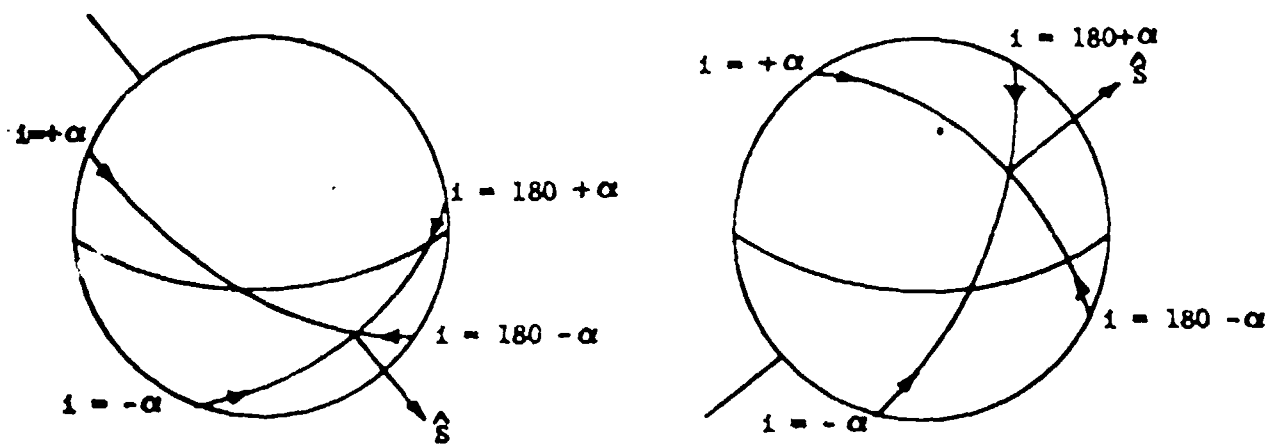


Figure 2. Possible Trajectories with Same Inclination

The detailed computations for the basic part of the program are straightforward. Using the standard conic abbreviations,

$$c = |\vec{r} \times \vec{v}| \tag{2}$$

$$\hat{w} = \frac{\vec{r} \times \vec{v}}{c} \tag{3}$$

$$p = \frac{c^2}{\mu} \tag{4}$$

$$a = \frac{r}{2 - rv^2/\mu} \tag{5}$$

$$e^2 = 1 - \frac{p}{a} \tag{6}$$

$$b = \sqrt{p\,|a|} \tag{7}$$

$$\cos f = \frac{p - r}{e\,r} \tag{8}$$

$$\sin f = \frac{\dot{r}\,c}{e\,\mu} \tag{9}$$

$$\hat{Z} = \frac{r}{c}\; v - \frac{\dot{r}}{c}\,\vec{r} \tag{10}$$

$$\hat{P} = \frac{\vec{r}}{r}\cos f - \vec{Z}\sin f \tag{11}$$

$$\hat{Q} = \frac{\vec{r}}{r}\sin f + \hat{Z}\cos f \tag{12}$$

$$\hat{S} = -\frac{a}{\sqrt{a^2+b^2}}\,\hat{P} + \frac{b}{\sqrt{a^2+b^2}}\,\hat{Q} \tag{13}$$

$$\hat{T} = \frac{\hat{S} \times \hat{K}}{\hat{S} \times \hat{K}} \tag{14}$$

$$\hat{R} = \hat{S} \times \hat{T} \tag{15}$$

$$\vec{B} = \frac{b^2}{\sqrt{a^2+b^2}}\,\hat{P} + \frac{ab}{\sqrt{a^2+b^2}}\,\hat{Q} \tag{16}$$

$$B \cdot T = \vec{B} \cdot \hat{T} \tag{17}$$

$$B \cdot R = \vec{B} \cdot \hat{R} \tag{18}$$

The computations for the optional part of the program which converts the i and r_{CA} into an equivalent $\vec{B}$ proceed as follows. The approach asymptote is first converted into target planet equatorial coordinates and its right ascension and declination computed

$$\hat{S}q = \phi_{ECEQ} \hat{S}$$

$$\theta_S = \tan^{-1} \frac{(Sq)_y}{(Sq)_x} \tag{19}$$

$$\delta_S = \sin^{-1} (Sq)_z$$

The angle $\Delta\theta$ between the ascending node of the trajectory and the right ascension of the approach asymptote is from Napiers rule

$$\sin \theta = \frac{\tan \delta_S}{\tan i} \tag{20}$$

after assuring that $|i| > |\delta_S|$. The ascending node of the trajectory is then computed recalling the definitions of the angle i

$$\Omega = \theta_S + \Delta\theta \quad (+\pi) \tag{21}$$

Thus the unit vector to the ascending node is given by

$$\hat{R}_A = (\cos\Omega, \sin\Omega, 0) \tag{22}$$

The normal to the orbital plane (in target planet equatorial coordinates) is

$$\hat{W}_q = \frac{\hat{S}q \times \hat{R}_A}{|\hat{S}q \times \hat{R}_A|} \tag{23}$$

This is now converted to the ecliptic coordinate system

$$\hat{W}_C = \phi^T_{ECEQ} \hat{W}q \tag{24}$$

The unit vector in the desired $\vec{B*}$ direction is

$$\hat{B*} = \frac{\hat{S} \times \hat{W}_C}{|\hat{S} \times \hat{W}_C|} \tag{25}$$

The magnitude of the $\vec{B^*}$ vector is given by

$$B^* = r_{CA}\sqrt{1 + \frac{2\mu}{r_{CA}\, V_\infty^2}} \tag{26}$$

Then the target impact parameter is $\vec{B^*} = B^* \widehat{B^*}$. The target values are then given by their obvious definitions

$$\begin{aligned} B\cdot T^* &= \vec{B^*} \cdot \hat{T} \\ B\cdot R^* &= \vec{B^*} \cdot \hat{R} \end{aligned} \tag{27}$$

Finally the hyperbolic time from $(\vec{r}, \vec{V})$ to periapsis is computed from the conic formula

$$\begin{aligned} \tanh \frac{F}{2} &= \sqrt{\frac{e-1}{e+1}}\, \tan \frac{f}{2} \\ t &= \sqrt{\frac{-a^3}{\mu}}\, (e \sinh F - F) \end{aligned} \tag{28}$$

Reference: Kizner, W., A Method of Describing Miss Distances for Lunar and Interplanetary Trajectories, Ballistic Missiles and Space Technology vol III, Pergamon Press, New York, 1961.

SUBROUTINE INPUTZ

PURPOSE: TO CONVERT THE INPUT INFORMATION FOR THE VIRTUAL MASS PROGRAM INTO VARIABLES COMPATIBLE WITH THE REST OF THE VIRTUAL MASS SUBROUTINES

CALLING SEQUENCE: CALL INPUTZ(RS,NTP,IPRINT)

ARGUMENTS

ARGUMENTS	RS(6)	I	INERTIAL STATE OF S/C AT INITIAL TIME
	NTP	I	CODE OF TARGET BODY
	IPRINT	I	INITIAL INFORMATION PRINT FLAG =0 PRINT INITIAL DATA =1 DO NOT PRINT INITIAL DATA

SUBROUTINES SUPPORTED: VMP

SUBROUTINES REQUIRED: TIME SPACE

LOCAL SYMBOLS:

D	INTERMEDIATE VARIABLE FOR PRINTOUT PURPOSES
D2	JULIAN DATE OF FINAL TRAJECTORY TIME
IDAY	DAY OF CALENDAR DATE OF FINAL TRAJECTORY TIME
IHR	HOUR OF CALENDAR DATE OF FINAL TRAJECTORY TIME
IMIN	MINUTE OF CALENDAR DATE OF FINAL TRAJECTORY TIME
IMO	MONTH OF CALENDAR DATE OF FINAL TRAJECTORY TIME
INERR	NOT USED
IP	CODE OF I-TH PLANET FOR STORAGE OF PMASS ARRAY
IYR	YEAR OF CALENDAR DATE OF FINAL TRAJECTORY TIME
LDAY	DAY OF CALENDAR DATE OF INITIAL TIME
LHR	HOUR OF CALENDAR DATE OF INITIAL TIME
LMIN	MINUTE OF CALENDAR DATE OF INITIAL TIME
LMO	MONTH OF CALENDAR DATE OF INITIAL TIME

LYR	YEAR OF CALENDAR DATE OF INITIAL TIME
SECI	SECOND OF CALENDAR DATE OF FINAL TIME
SECL	SECOND OF CALENDAR DATE OF INITIAL TIME
TP	INTERMEDIATE VARIABLE FOR CALCULATION OF COMPUTING INTERVAL

COMMON COMPUTED/USED: V

COMMON COMPUTED: F INC IPR ITRAT KOUNT NBODY

COMMON USED: NBODYI NO PMASS ZERO

SUBROUTINE INSERS

PURPOSE: TO CONTROL THE PROCESSING OF AN ORBITAL INSERTION EVENT.

CALLING SEQUENCE: CALL INSERS(DTIME)

ARGUMENT: DTIME I TIME INTERVAL FROM DECISION TO EXECUTION (DAYS)

SUBROUTINES SUPPORTED: GIDANS

SUBROUTINES REQUIRED: COPINS NONINS PECEQ

LOCAL SYMBOLS:

DA	DESIRED SEMIMAJOR AXIS
DE	DESIRED ECCENTRICITY
DI	DESIRED INCLINATION
DN	DESIRED LONGITUDE OF ASCENDING NODE
DWTP	DESIRED ARGUMENT OF PERIAPSIS SHIFT OR DESIRED ARGUMENT OF PERIAPSIS
ECEQI	ECLIPTIC TO EQUATORIAL TRANSFORMATION
GM	GRAVITIONAL CONSTANT OF TARGET BODY
IEX	UNEXECUTABLE EVENT CODE =0 EVENT IS EXECUTABLE =1 NO EXECUTABLE SOLUTION FOUND
IOPT	INSERTION STRATEGY OPTION =1 COPLANAR INSERTION =2 NONPLANER INSERTION
RSP	SPACECRAFT POSITION IN ECLIPTIC COORDS
RSQ	SPACECRAFT POSITION IN EQUATORIAL COORDS
TEX	TIME INTERVAL TO EXECUTION (SECONDS)
VSP	SPACECRAFT VELOCITY IN ECLIPTIC COORDS
VSQ	SPACECRAFT VELOCITY IN EQUATORIAL COORDS

COMMON COMPUTED/USED: DELTAV

COMMON COMPUTED: DELV KTIM KWIT

COMMON USED: ALNGTH D1 F KMXQ KTAR KUR NBOD NB NTP PMASS

TAR TN V

INSERS Analysis

INSERS controls the processing of an orbital insertion event. The subroutine COPINS and NONINS perform the actual computations for the coplanar and non-planar options respectively.

INSERS first records the specific parameter ...ues for the current orbit insertion event.

It then computes the current state ($\vec{r}$, $\vec{v}$) of ... spacecraft in target-planet centered ecliptic coordinates. Subroutine PECEQ is called to compute the transformation matrix Φ_{ECEQ} from ecliptic to equatorial coordinates. The planet centered equatorial coordinates are then

$$\vec{r}_q = \Phi_{ECEQ}\,\vec{r}$$

$$\vec{v}_q = \Phi_{ECEQ}\,\vec{v}$$

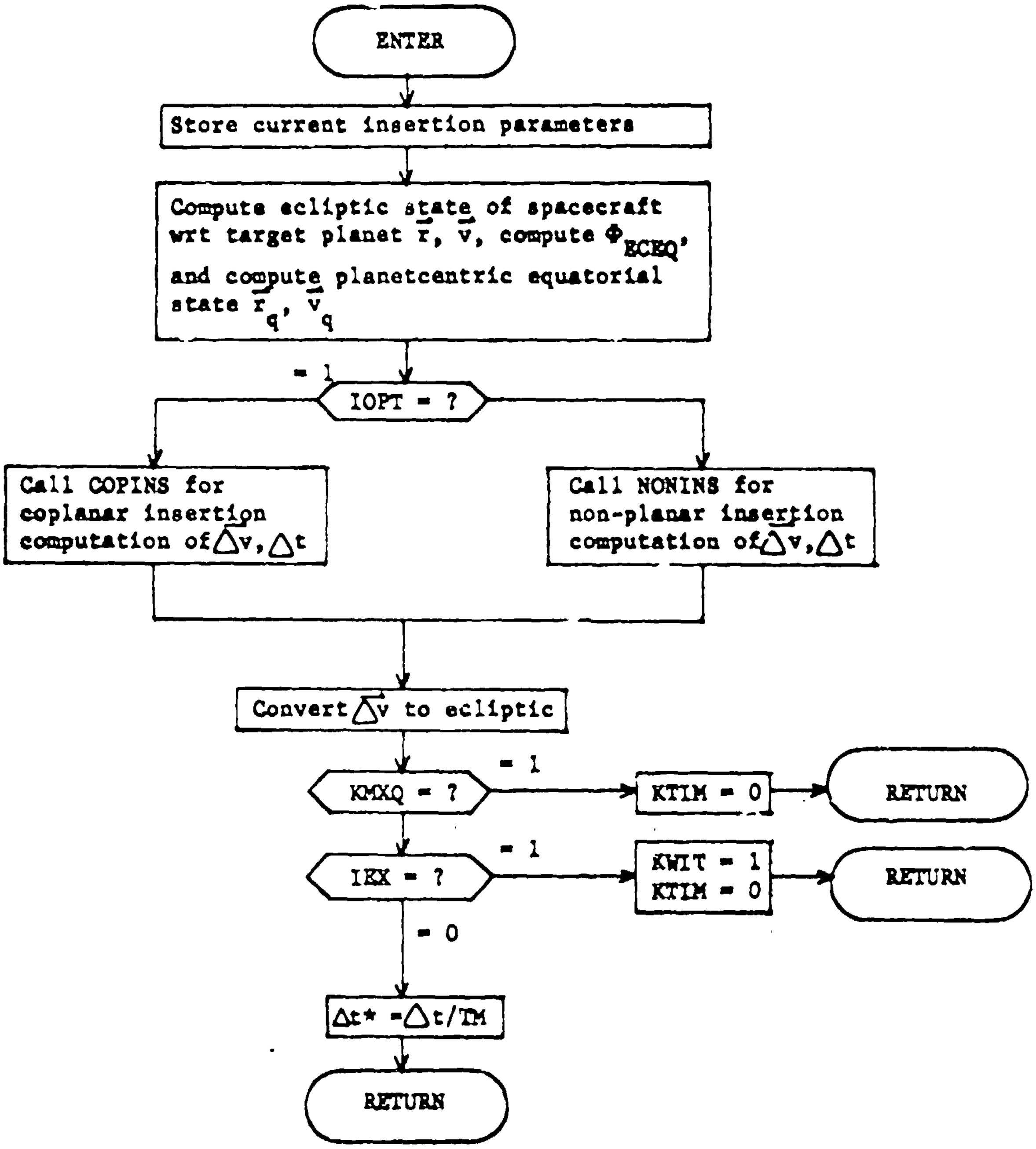
This state is then sent to COPINS or NONINS for the computation of the insertion velocity Δv_q and the time interval t between the current time and the time at which the insertion should take place (based on conic propagation about the target body). The correction Δv_q is then converted to ecliptic coordinates

$$\overrightarrow{\Delta v} = \Phi^T\,\overrightarrow{\Delta v_q}$$

If the event is a compute-only mode, the return is made to GIDANS.

If the event is to be executed the flag IEX (set by COPINS or NONINS to indicate success or failure) is then interrogated. If IEX = 1, no acceptable insertion event was found and so the executive flag KWIT is set to 1 before returning. If IEX = 0 an acceptable insertion was determined and so it is set up.

INSERS Flow Chart



SUBROUTINE JACOBI

PURPOSE: TRANSFORMATION OF A REAL SYMMETRIC MATRIX TO DIAGONAL FORM BY A SUCCESSION OF PLANE ROTATIONS TO ANNIHILATE THE OFF-DIAGONAL ELEMENTS AND SUBSEQUENT COMPUTATION OF THE EIGENVALUES AND EIGENVECTORS OF THAT MATRIX

CALLING SEQUENCE: CALL JACOBI(A,W2,V,N,FOD)

ARGUMENTS:

ARGUMENTS:	A	I	MATRIX TO BE DIAGONALIZED (WILL BE DESTROYED)
	W2	O	VECTOR OF EIGENVALUES (LENGTH N)
	V	O	MATRIX OF EIGENVECTORS (N BY N DIMENSION)
	N	I	DIMENSION OF SQUARE MATRIX A
	FOD	I	FINAL OFF-DIAGONAL ANNIHILATION VALUE

SUBROUTINES SUPPORTED: EIGHY GUISIN GUISS PRESIN SETEVN GUIDM GUID PRED

LOCAL SYMBOLS:

AIIP	INTERMEDIATE VARIABLE
AIPIP	INTERMEDIATE VARIABLE-A(IPIP)
AIPJP	INTERMEDIATE VARIABLE-A(IPJP)
AJPJP	INTERMEDIATE VARIABLE-A(JPJP)
CS	INTERMEDIATE VARIABLE
DEL	DIFFERENCE IN ELEMENTS OF A
IREDO	COUNTER
KR	DIMENSION OF A
KRP1	KR + 1
NM1	N - 1
RAD	INTERMEDIATE VARIABLE
SN	INTERMEDIATE VARIABLE
TN	INTERMEDIATE VARIABLE
T1	LARGEST OFF-DIAGONAL ELEMENT
VIIP	INTERMEDIATE VARIABLE

COMMON USED: ONE TWO ZERO

JACOBI Analysis

The Jacobi method subjects a real, symmetric matrix A to a sequence of transformations based on a rotation matrix:

$$O_K = \begin{bmatrix} \cos\phi_K & -\sin\phi_K \\ \sin\phi_K & \cos\phi_K \end{bmatrix}$$

where all other elements of the rotation matrix are identical with the unit matrix. After n multiplications A is transformed into:

$$A' = O_N^{-1} \ldots . O_1^{-1} A\, O_1 \ldots . O_N$$

If ϕ_K is chosen at each step to make a pair of off-diagonal elements zero, then A' will approach diagonal form with the eigenvalues on the diagonal. The columns of $O_1\, O_2 \ldots O_N$ correspond to the eigenvectors of A.

The angle of rotation ϕ is chosen in the following way. If the four entries of O_K are in (i,i), (i,j), (j,i) and (j,j) then the corresponding elements of $O_1^{-1} A\, O_1$ are

$$b_{ii} = a_{ii} \cos^2\phi + 2a_{ij} \sin\phi \cos\phi + a_{jj} \sin^2\phi$$

$$b_{ij} = b_{ji} = (a_{jj} - a_{ii}) \sin\phi \cos\phi + a_{ij}(\cos^2\phi - \sin^2\phi)$$

$$b_{jj} = a_{ii} \sin^2\phi - 2a_{ij} \sin\phi \cos\phi + a_{jj} \cos^2\phi$$

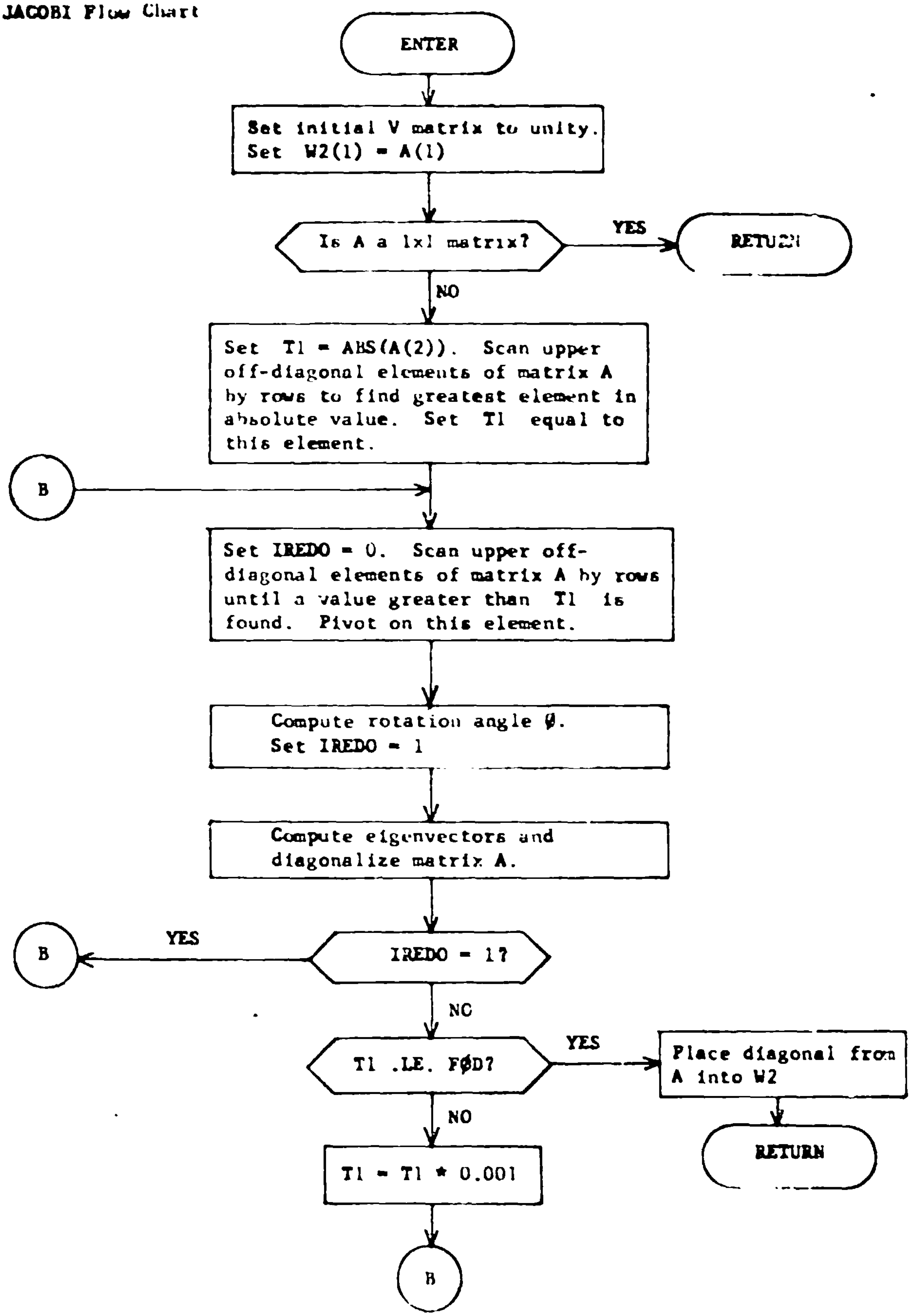
If ϕ is chosen so that $\tan 2\phi = 2a_{ij}/(a_{ii} - a_{jj})$ then

$$b_{ij} = b_{ji} = 0$$

Each multiplication creates a new pair of zeros but will introduce a non-zero contribution to positions zeroed out on previous steps. However, successive matrices of the form $O_2^{-1}\, O_1^{-1} A\, O_1\, O_2$ will approach the required diagonal form.

Reference: Scheid, Frances: Theory and Problems of Numerical Analysis, McGraw-Hill Book Company, Inc., New York, 1968.

JACOBI Flow Chart



SUBROUTINE LAUNCH

PURPOSE: TO COMPUTE THE INJECTION TIME, POSITION, AND VELOCITY FROM THE DEPARTURE ASYMPTOTE AND THE LAUNCH PROFILE

ARGUMENT:	DI	JULIAN DATE AT INJECTION (OUTPUT)
	RZ	POSITION VECTOR AT INJECTION (OUTPUT)
	VZ	VELOCITY VECTOR AT INJECTION (OUTPUT)

SUBROUTINES SUPPORTED: HELIO

SUBROUTINES REQUIRED: EPHEM ORB PECEQ

LOCALS SYMBOLS:	ANGLE	INTERMEDIATE ANGLE USED TO DEFINE TL
	AZI	PLANETOCENTRIC AZUMUTH AT INJECTION (DEG)
	BHAT	UNIT VECTOR NORMAL TO SHAT AND WHAT USED TO DEFINE THE P-Q ELEMENTS OF THE DEPARTURE HYPERBOLA
	BMAG	MAGNITUDE OF THE NON-UNITIZED BHAT VECTOR
	COSFL	COSINE OF FL
	COSFS	COSINE OF FS
	COSGAM	COSINE OF GAMMAI
	COSPHI	COSINE OF FI
	COSSIG	CONSINE OF SIGMAL
	COSWL	CONSINE OF WL
	C3	VIS VIVA ENERGY ON THE DEPARTURE HYPERBOLA
	DD	INTERMEDIATE VARIABLE USED TO CALCULATE GREENWICH HOUR ANGLE
	DLA	PLANETOCENTRIC EQUATORIAL DECLINATION OF THE DEPARTURE ASYMPTOTE
	EQEC	TRANSFORMATION MATRIX FROM ECLIPTIC TO LAUNCH PLANET EQUATORIAL
	FL	TRUE ANOMALY OF LAUNCH SITE POSITION VECTOR

FS	TRUE ANOMALY OF DEPARTURE ASYMPTOTE
GAMMAI	FLIGHT PATH ANGLE AT INJECTION
GH	GREENWICH HOUR ANGLE
GMLP	GRAVITATIONAL CONSTANT OF THE LAUNCH PLANET IN KM**3/SEC**2
HE	ECCENTRICITY OF THE DEPARTURE HYPERBOLA
ID	INTERMEDIATE VARIABLE USED TO COMPUTE GREENWICH HOUR ANGLE
IHR	HOUR OF INJECTION
IMN	MINUTE OF INJECTION
I	INDEX
J	INDEX
LHR	HOUR OF LAUNCH
LMN	MINUTE OF LAUNCH
PHAT	UNIT VECTOR POINTING TOWARD PERIAPSIS OF THE HYPERBOLA
PHII	LATITUDE OF INJECTION
PSIB	THE ANGLE FROM LAUNCH TO INJECTION
QHAT	UNIT VECTOR NORMAL TO PHAT POINTING IN THE DIRECTION OF MOTION
RAI	RIGHT ASCENSION AT INJECTION
RAL	RIGHT ASCENSION OF DEPARTURE ASYMPTOTE
REFJD	JULIAN DATE FOR 1950
RIMAG	MAGNITUDE OF THE SPACECPAFT PCSITION AT INJECTION
RI	SPACECRAFT POSITION AT INJECTION
RLHAT	LAUNCH SITE POSITION UNIT VECTOR
SECI	SECOND OF INJECTION
SECL	SECOND OF LAUNCH

SHAT	UNIT SPACECRAFT VELOCITY VECTOR IN EQUATORIAL SYSTEM AT INJECTION
SINFL	SINE OF FL
SINFS	SINE OF FS
SINGAM	SINE OF GAMMAI
SINPHI	SINE OF FI
SINSIG	SINE OF SIGMAL
SINWL	SINE OF WL
SLR	SIMI-LATUS RECTUM OF THE DEPARTURE HYPERBOLA
TB	TIME BETWEEN LAUNCH AND INJECTION IN SECONDS
TC	LENGTH OF PARKING ORBIT COAST IN SECONDS
TEST	INTERMEDIATE VARIABLE TO TEST FOR VIOLATION OF AZIMUTH CONSTRAINT
TFRAC	INTERMEDIATE VARIABLE USED TO CALCULATE GREENWICH HOUR ANGLE
THETAI	LONGITUDE AT INJECTION
TH	INTERMEDIATE VARIABLE USED TO CALCULATE CLOCK TIMES OF LAUNCH AND INJECTION
TI	INJECTION TIME IN DAYS REFERENCED TO MIDNIGHT OF THE LAUNCH DAY
TL	LAUNCH TIME IN DAYS REFERENCED TO MIDNIGHT OF THE LAUNCH DAY
TMN	INTERMEDIATE VARIABLE USED TO CALCULATE CLOCK TIMES OF LAUNCH AND INJECTION
TSTAR	INTERMEDIATE VARIABLE USED TO COMPUTE GREENWICH HOUR ANGLE
TWOFOR	CONSTANT VALUE, EQUAL TO 24.
VHL	MAGNITUDE OF VZ, THE INPUT VECTOR OF THE DEPARTURE ASYMPTOTE
VIMAG	MAGNITUDE OF SPACECRAFT VELOCITY AT INJECTION

WHAT	UNIT VECTOR NORMAL TO THE LAUNCH PLANE IN EQUATORIAL SYSTEM
WL	RIGHT ASCENSION OF THE LAUNCH SITE
WMAG	MAGNITUDE OF THE NON-UNITIZED WHAT VECTOR
XTIM	INTERMEDIATE VARIABLE USED TO COMPUTE CLOCK TIMES OF LAUNCH AND INJECTION

COMMON COMPUTED/USED: SIGMAL

COMMON COMPUTED: NO

COMMON USED:

ALNGTH	DPA	FI	FOUR	KOAST
NINETY	NLP	ONE	PHILS	PMASS
PSI1	PSI2	RAD	RAP	RPRAT
RP	THEDOT	THELS	TIM1	TIM2
TM	TWO	VHPM	XP	ZERO

LAUNCH Analysis

LAUNCH computes the injection time, position and velocity from the departure velocity $\vec{v}_{HE}$ (computed in HELIO) and the launch profile parameters input by the user.

The rotation matrix Φ_{ECEQ} defining the transformation from ecliptic to equatorial coordinates is first computed (PECEQ). The departure velocity $\vec{v}_{HE}$ is then normalized and converted into ecliptic coordinates to yield the departure asymptote $\hat{S}$.

$$\hat{S} = \Phi_{ECEQ} \frac{\vec{v}_{HE}}{v_{HE}} \tag{1}$$

Auxiliary information associated with $\hat{S}$ is then computed. The energy C_3, the declination Φ_S and the right ascension θ_S of the departure asymptote, and the eccentricity of the departure hyperbola are given by

$$C_3 = v_{HE}{}^2$$

$$\sin \Phi_S = S_z$$

$$\tan \theta_S = \frac{S_y}{S_x}$$

$$e = 1 + \frac{r_p C_3}{\mu} \tag{2}$$

where r_p is the desired parking orbit radius and μ is the gravitational constant of the launch planet.

The unit normal $\hat{W}$ to the launch plane in equatorial coordinates is then computed. $\hat{W}$ is defined by

$$W_z = \cos \Phi_L \sin \Sigma_L$$

$$W_y = \frac{-W_z S_y S_z + k S_x \left[1 - (S_z^2 + W_z^2)\right]^{\frac{1}{2}}}{S_x^2 + S_y^2}$$

$$W_x = \frac{-(W_y S_y + W_z S_z)}{S_x} \tag{3}$$

where Φ_L is the launch site latitude, Σ_L is the launch azimuth, and $k = +1$ or -1 for the long or short coast time models respectively. The second equation defines an implicit constraint on Σ_L

$$\sin^2 \Sigma_L \leq \frac{\cos^2 \Phi_s}{\cos^2 \Phi_L} \tag{4}$$

The right ascension at launch θ_L may now be defined by

$$\cos \theta_L = \frac{W_x \sin \Phi_L \sin \Sigma_L + W_y \cos \Sigma_L}{W_z^2 - 1}$$

$$\sin \theta_L = \frac{W_y \sin \Phi_L \sin \Sigma_L - W_x \cos \Sigma_L}{W_z^2 - 1} \tag{5}$$

and the unit vector toward the launch position is then

$$R_L = (\cos \Phi_L \cos \theta_L \ , \cos \Phi_L \sin \theta_L \ , \sin \Phi_L) \tag{6}$$

The complementary unit vectors $\hat{P}$, $\hat{Q}$ defining the orientation of the hyperbola within the launch plane are now introduced. Let

$$\hat{B} = \hat{S} \times \hat{W} \tag{7}$$

The true anomaly of the departure asymptote is $\cos f_s = -\frac{1}{e}$. Then $\hat{P}$ and $\hat{Q}$ are given as

$$\hat{P} = \hat{S} \cos f_s + \hat{B} \sin f_s$$
$$\hat{Q} = \hat{S} \sin f_s - \hat{B} \cos f_s \tag{8}$$

The true anomaly of the launch site f_I may now be given

$$\cos f_L = \hat{R}_L \cdot \hat{P}$$
$$\sin f_L = \hat{R}_L \cdot \hat{Q} \tag{9}$$

The angle Ψ_B between launch and injection is

$$\Psi_B = 2\pi - f_L + f_I \tag{10}$$

where f_I is the desired true anomaly at injection read in as input. The coast time t_c may now be computed from

$$t_c = \left[\Psi_B - (\Psi_1 + \Psi_2)\right] k_{\dot{\Phi}} \qquad (11)$$

where Ψ_1 and Ψ_2 are the angles of the first and second burns and $k_{\dot{\Phi}}$ is the inverse of the parking orbit coast rate, all of which are read in as input.

The time between launch and injection is therefore

$$t_B = t_1 + t_2 + t_c \qquad (12)$$

where t_1 and t_2 are the input time durations of the first and second burns.

The unit vector to injection is

$$\hat{R}_I = \hat{P} \cos f_I + \hat{Q} \sin f_I \qquad (13)$$

The semi-latus rectum p is

$$p = \frac{\mu(e^2 - 1)}{C_3} \qquad (14)$$

The radius magnitude to injection is

$$R_I = \frac{p}{1 + e \cos f_I} \qquad (15)$$

The injection speed is

$$V_I = \sqrt{C_3 + \frac{2\mu}{R_I}} \qquad (16)$$

The path angle at injection is

$$\cos \Gamma_I = \frac{\sqrt{\mu p}}{R_I V_I} \qquad (17)$$

The injection latitude is

$$\sin \Phi_I = \hat{R}_{I_z} \tag{18}$$

The injection right ascension is

$$\tan \Theta_I = \frac{R_{I_y}}{R_{I_x}} \tag{19}$$

The injection longitude is

$$\theta_I = \theta_L + \Theta_I - \Theta_L - \omega t_B \tag{20}$$

where θ_L is the longitude of the launch site and ω is the rotation rate of the launch planet, both being read in as input.

The injection azimuth is

$$\cos \Sigma_I = \frac{S_z - \cos (f_s - f_I) \sin \Phi_I}{\sin (f_s - f_I) \cos \Phi_I} \tag{21}$$

The launch time on the day of launch is

$$t_L = \frac{(\Theta_L - \theta_L - GHA) \bmod 2\pi}{\omega} \tag{22}$$

where GHA is the Greenwich hour angle at 0^h UT of the launch date

$$GHA = 100^{\circ}.07554260 + 0^{\circ}.9856473460\, T_d + 2^{\circ}.9015 \times 10^{-3}\, T_d^{\,3} \tag{23}$$

where T_d = days past 0^h January 1, 1950.

The injection radius vector is now computed from

$$\vec{R}_I = R_I \hat{R}_I \tag{24}$$

$$\vec{V}_I = \frac{V_I}{R_I} \left[(\hat{W} \times \vec{R}_I) \cos \Gamma_I + \vec{R}_I \sin \Gamma_I \right]$$

The injection time is

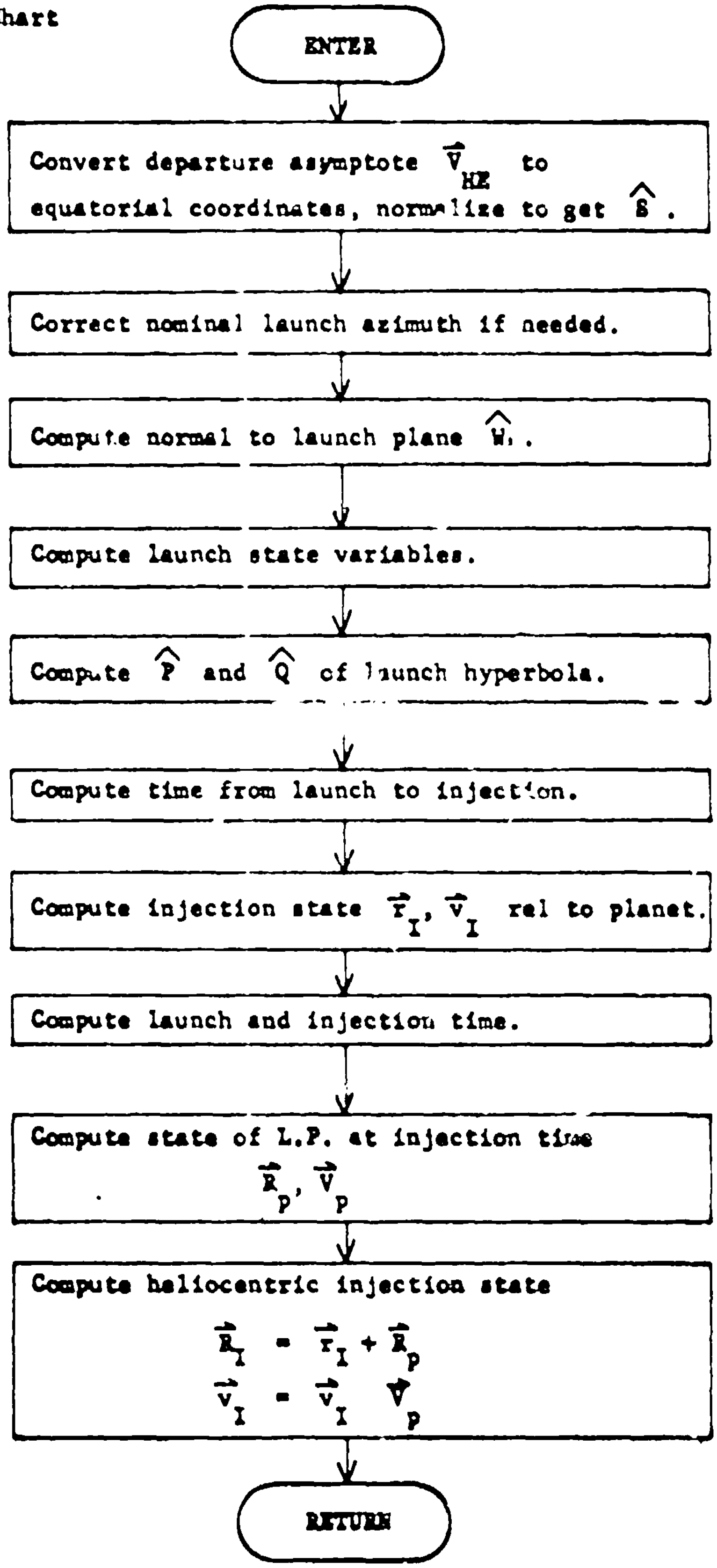
$$T_I = T_o + t_L + t_B \tag{25}$$

where T_o is the Julian date of the launch calendar date.

The injection position and velocity are now rotated into the ecliptic plane. The position and velocity of the launch planet at the time T_I are computed and added to the injection state to get the heliocentric injection state.

Reference: Space Research Conic Program, Phase III, May 1, 1969, Jet Propulsion Laboratory, Pasadena, California.

LAUNCH Flow Chart



SUBROUTINE LUNA

PURPOSE: TO CONTROL THE GENERATION OF THE ZERO ITERATE FOR LUNAR TARGETING

CALLING SEQUENCE: CALL LUNA

SUBROUTINES SUPPORTED: ZERIT

SUBROUTINES REQUIRED: LUNTAR MULTAR

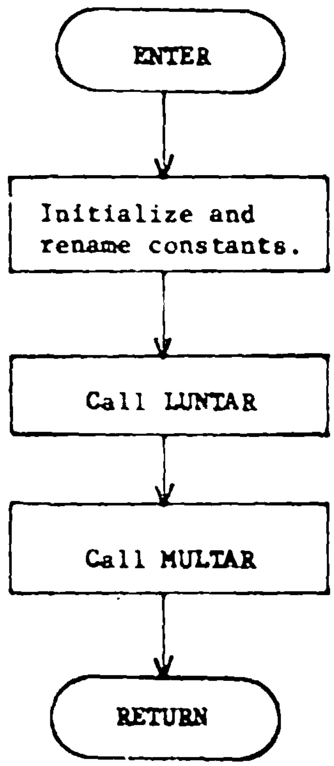
LOCAL SYMBOLS:	I	INDEX
	OSPH	ORIGINAL SPHERE OF INFLUENCE OF TARGFT PLANET IN A.U.

COMMON COMPUTED/USED:	OTAR	SPHERE			
COMMON COMPUTED:	BCON	CAI	IBARY	ICOORD	PCON
	RCA	RPE	SMA	TCA	TSPH
	TTOL				
COMMON USED:	ALNGTH	DT	FOUR	KUR	NTP
	ONE	RP	SPHFAC	TEN	ZDAT

LUNA Analysis

LUNA is the controlling subroutine for lunar zero iterate targeting. It first serves an interface role in which it initializes constants and renames variables for the other lunar targeting routines. It then calls LUNTAR for the targeting of the lunar patched conic. When that is completed it calls MULTAR for the targeting of the multi conic trajectory. It then returns control to PRELIM.

LUNA Flow Chart



SUBROUTINE LUNCON

PURPOSE: TO COMPUTE THE ACTUAL VALUES OF THE TARGET PARAMETERS (A, BDT, BDR) FOR A LUNAR PATCHED CONIC TRAJECTORY DETERMINED BY CONTROL VALUES OF ALPHA, DELTA, AND THETA.

CALLING SEQUENCE: CALL LUNCON(ALPHAI,DELTAI,THETAI,AM,BDT,BDR,
SIGNAL,ITR)

ARGUMENTS:

ALPHAI	I	ANGLE DEFINING PERIGEE OF TRANSFER CONIC (RAD)
DELTAI	I	DELINATION OF LSI POINT (RAD)
THETAI	I	RIGHT ASCENSION OF LSI POINT (RAD)
AM	O	SEMIMAJOR AXIS OF LUNAR CONIC
BDT	O	IMPACT PARAMETER OF LUNAR CONIC
BDR	O	IMPACT PARAMETER OF LUNAR CONIC
SIGNAL	I/O	NOMINAL LAUNCH AZIMUTH OR THAT REQUIRED
ITR	O	OUTPUT ITERATION COUNTER

SUBROUTINES SUPPORTED: LUNTAR

SUBROUTINES REQUIRED: CAREL IMPACT

LOCAL SYMBOLS:

ALPHA	ALPHAI IN DEGREES
AOUT	TEMPORARY LOCATION FOR AM
CC	ANGULAR MOMENTUM OF THE EARTH CENTERED TRANSFER CONIC
CDEL	COSINE OF DELTAI
CECC	ECCENTRICITY OF THE EARTH CENTERED TRANSFER CONIC
COSDEC	COSINE OF DECLIN
COSPL	COSINE OF PHIL
COSPS	INTERMEDIATE VARIABLE TO TEST FOR VIOLATIN OF SIGNAL CONSTAINT
COSSIG	COSINE OF SIGNAL
CP	SEMI-LATUS RECTUM OF EARTH CENTERED TRANSFER CONIC

CSMA	SEMI-MAJOR AXIS OF EARTH CENTERED TRANSFER CONIC
CT	COSINE OF THETAI
DELTA	DELTAI IN DEGREES
EM	ECCENTRICITY OF LUNAR CONIC
GAMMI	INTERMEDIATE ANGLE USED TO COMPUTE EARTH CENTERED TRANSFER CONIC
I	INDEX
PHIL	LATITUDE OF LAUNCH SITE
POS	SPACECRAFT POSITION AND VELOCITY AT LSI POINT IN M ~-CENTERED EARTH EQUATORIAL COORDINATES
PPM	DUMMY VARIA LE FOR CALL CAREL
QQM	DUMMY VARIABLE FOR CALL TO CAREL
RAD	RADIANS TO DEGREES CONVERSION FACTOR
RMAG	MAGNITUDE OF THE RI VECTOR
ROUT	VELOCITY AT LSI IN GEOCENTRIC EQUATORIAL SYSTEM
RPM	RADIUS OF PERIAPSIS OF LUNAR CONIC
SDEL	SINE OF DELTAI
SHAT	UNIT VECTOR POINTING FROM THE EARTH TO THE POINT DEFINED BY DELTAI, THETAI
SIGM	SIGMAL IN DEGREES
SINDEC	SINE OF DECLIN
SINPS	INTERMEDIATE VARIABLE USED TO TEST FOR VIOLATION OF SIGMAL CONTRAINT
SINSIG	SINE OF SIGMAL
SX	SINE OF THETAI
TAM	TRUE ANOMALY OF THE LUNAR CONIC CORRESPONDING TO THE RSI VECTOR

TFLP	TIME OF FLIGHT FROM PERIAPSIS CORRESPOND-ING TO THE RSI VECTOR
THETA	THETAI IN DEGREES
VMAG	SPACECRAFT VELOCITY MAGNITUDE USED TO CALCULATE DECLIN
WHAT	UNIT VECTOR NORMAL TO THE EARTH-PHASE
WMAG	ANGULAR MOMENTUM CONSTANT
WM	ARGUMENT OF PERIAPSIS OF THE LUNAR CONIC
WWM	DUMMY VARIABLE FOR CALL TO CAREL
XHAT	SAME AS SHAT VECTOR
XIM	INCLINATION OF THE CONIC
XNM	LONGITUDE OF THE ASCENDING NODE OF THE LUNAR CONIC
YHAT	CROSS PRODUCT OF THE WHAT AND XHAT VECTORS

COMMON COMPUTED/USED: RI RSI

COMMON COMPUTED: DECLIN

COMMON USED:

EMU	EQLQ	KOAST	NINETY	ONE
PHILS	RMQ	RPE	SIGMA	TMU
TSPH	TWO	ZERO		

LUNCON Analysis

The point of intersection of the Earth-centered conic with the lunar sphere of influence (LSI) is determined by the angles and δ. Relative to the moon in Earth-equatorial coordinates that point is

$$\vec{r_{SI}} = \begin{bmatrix} R_{SI} \cos\delta \cos\theta \\ R_{SI} \cos\delta \sin\theta \\ R_{SI} \sin\delta \end{bmatrix} \quad (1)$$

where R_{SI} is the radius of the LSI. Relative to the earth that point is

$$\vec{R_q} = \vec{R_M} + \vec{r_{SI}} \quad (2)$$

where $\vec{R_M}$ is the radius vector to the center of the moon at the time of LSI intersection t_{SI} in earth equatorial coordinates.

There are at most two planes which contain $\vec{R_q}$ and satisfy the launch latitude ϕ and azimuth Σ constraints. Let $\hat{W}$ denote the unit normal to either of these planes. Now let $\hat{R}_L$, θ_L, ϕ_L denote the unit vector, longitude, and latitude of the launch site. Construct a local horizon coordinate system at the launch site as indicated in Figure 1.

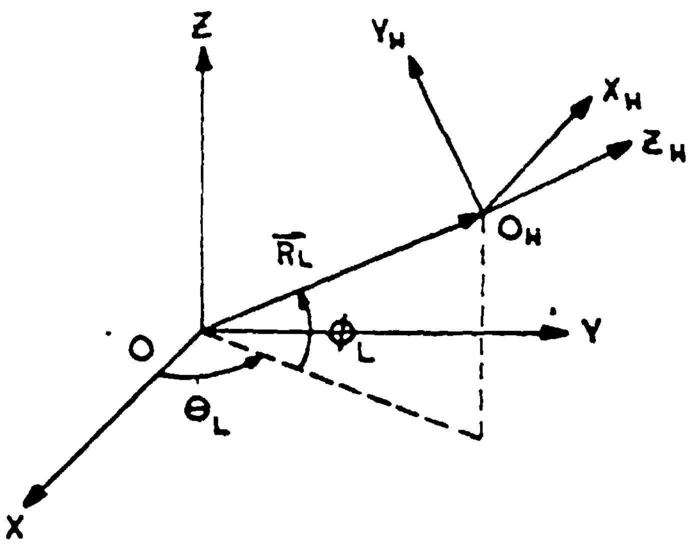


Figure 1. Local Horizon Coordinate System

Here $\hat{Z}_h = \frac{\hat{R}_L}{R_L}$, $\hat{Y}_L$ is normal to $\hat{Z}_h$ in the $\hat{Z}$-O-$\hat{R}_L$ plane, and $\hat{X}_h = \hat{Y}_h \times \hat{Z}_h$.

In the local horizon system, the position and velocity are very simply represented

$$\vec{R}_h = R\,[0,\ 0,\ 1]^T$$
$$\vec{V}_h = V\,[\cos\delta\ \sin\Sigma\ ,\ \cos\delta\ \cos\Sigma\ ,\ \sin\delta]^T \tag{3}$$

where Σ is the launch azimuth and δ is the declination wrt the local horizontal. Thus

$$\hat{W}_h = \frac{\vec{R}_h \times \vec{V}_h}{|\vec{R}_h \times \vec{V}_h|} = \begin{bmatrix} -\cos\Sigma \\ \sin\Sigma \\ 0 \end{bmatrix} \tag{4}$$

The transformation matrix converting a vector in the local horizon system to the equatorial system is

$$T = \begin{bmatrix} -\sin\theta_L & -\sin\phi_L\cos\theta_L & \cos\phi_L\cos\theta_L \\ \cos\theta_L & -\sin\phi_L\cos\theta_L & \cos\phi_L\sin\theta_L \\ 0 & \cos\phi_L & \sin\phi_L \end{bmatrix} \tag{5}$$

Therefore since $\vec{W}_q = T\,\vec{W}_h$, the z-component of $\hat{W}$ in the equatorial coordinate system is

$$\hat{W}_z = \cos\phi_L\ \sin\Sigma \tag{6}$$

Since $\hat{W}$ is a unit normal it must satisfy both $\hat{W}\cdot\hat{W} = 1$ and $\hat{W}\cdot\hat{S} = 0$ where $\hat{S} = \frac{\hat{R}_q}{R_q}$. Solving for the two remaining components of $\hat{W}$,

$$\hat{W}_y = \frac{-\hat{W}_z\ \hat{S}_y\ \hat{S}_z \pm \hat{S}_x\sqrt{1 - (\hat{S}_z^2 + \hat{W}_z^2)}}{\hat{S}_x^2 + S_y^2} \tag{7}$$

$$\hat{W}_x = \frac{-(\hat{W}_y\ \hat{S}_y + \hat{W}_z\ \hat{S}_z)}{\hat{S}_x} \tag{8}$$

To eliminate the ambiguity of sign in (7) the short-coast plane corresponding to the negative sign is used. Note that (7) also imposes a constraint on the launch azimuth

$$\sin^2 \Sigma \leqslant \frac{1 - \hat{S}_z^2}{\cos^2 \phi_L} \tag{9}$$

Now choose $\hat{U} = \hat{W} \times \hat{S}$ to complete a right hand system $(\hat{S}, \hat{U}, \hat{W})$. Then the position at LSI relative to the earth is $(R_I, 0, 0)$. Now let α determine the perigee point in the orbital plane $(\hat{W} = 0)$ measured counterclockwise from the $-\hat{S}$ axis. Then the perigee point is $(-r_p \cos\alpha, -r_p \sin\alpha, 0)$ where r_p is the parking orbit radius (input). Therefore the true anomaly of the earth centered conic at the LSI is given by

$$f_{SI} = 180 - \alpha \tag{10}$$

The two equations $R_I = \dfrac{a(1 - e^2)}{1 + e \cos f_{SI}}$ and $r_p = a(1 - e)$ may be solved simultaneously for the semi-major axis a and eccentricity e of the unique earth centered conic

$$e_g = \frac{R_I - r_p}{r_p - R_I \cos f_{SI}} \tag{11}$$

$$a_g = \frac{r_p}{1 - e_g} \tag{12}$$

Thus the velocity of the earth centered conic at the LSI is in the $(\hat{S}, \hat{U}, \hat{W})$ system

$$\vec{V}_o = \begin{bmatrix} \sqrt{a(1-e^2)} \;\; e \sin f_{SI} \\ \mu a(1-e^2) / R_I \\ 0 \end{bmatrix} \tag{13}$$

Transforming to the earth equatorial coordinate system

$$\vec{V}_q = \begin{bmatrix} S_x & U_x & W_x \\ S_y & U_y & W_y \\ S_z & U_z & W_z \end{bmatrix} \vec{V}_o \tag{14}$$

Now if $(\overrightarrow{R_{MQ}}, \overrightarrow{V_{MQ}})$ are the position and velocity of the moon at t_{SI} Earth-centered coordinates and $(\overrightarrow{R_Q}, \overrightarrow{V_Q})$ are the position and velocity of the spacecraft at t_{SI} then the state of the spacecraft with respect to the moon at t_{SI} is in earth equatorial coordinates

$$\begin{aligned} \overrightarrow{r_{SI}} &= \overrightarrow{R_Q} - \overrightarrow{R_{MQ}} \\ \overrightarrow{v_{SI}} &= \overrightarrow{V_Q} - \overrightarrow{V_{MQ}} \end{aligned} \tag{15}$$

Using the transformation matrix ϕ_{EQLQ} defining transformations from earth equatorial to lunar equatorial the state in the LQ system is

$$\begin{aligned} \overrightarrow{r_{sq}} &= \phi_{EQLQ}\,\overrightarrow{r_{SI}} \\ \overrightarrow{v_{sq}} &= \phi_{EQLQ}\,\overrightarrow{v_{SI}} \end{aligned} \tag{16}$$

The impact plane parameters B·T and B·R, and the inclination i_ℓ, may now be computed by calling subroutines ACTB and CAREL.

SUBROUTINE LUNTAR

PURPOSE: TO GENERATE A PATCHED CONIC TRAJECTORY FOR LUNAR MISSIONS CONSISTENT WITH TARGET PARAMETERS AT THE MOON OF (ACA, RCA, ICA, TCA) AND LAUNCH PARAMETERS (PHIL, THETAL, SIGMAL).

CALLING SEQUENCE: CALL LUNTAR

SUBROUTINES SUPPORTED: LUNA

SUBROUTINES REQUIRED: LUNCON EPHEM IMPACT MATIN ORB PECEQ

LOCAL SYMBOLS:

AA	SEMI-MAJOR AXIS OF THE LUNAR CONIC FOR THE NOMINAL TRAJECTORY
ALNGTH	SAME AS AU
ALPHAI	REFINED ANGLE (RADIANS) DEFINING POSITION OF PERIGEE ON THE TRANSFER CONIC (NOMINALLY SET TO FIVE DEGREES)
ALPI	PERTURBED VALUE OF ALPHAI USED TO SOLVE FOR RCA, ICA, ACA
AUDAY	CONVERTS KM/SEC TO AU/DAY
AUS	SAME AS AU
AU	CONVERTS KILOMETERS (KM) TO ASTRONOMICAL UNITS (AU)
BDR	B DOT R FOR THE NOMINAL TRAJECTORY
BDT	B DOT T FOR THE NOMINAL TRAJECTORY
BINC	OBTAINABLE INCLINATION USED TO CALCULATE DESIRED B DOT T, B DOT R
DELI	PERTURBED VALUE OF DELTAI USED TO SOLVE FOR RCA, ICA, ACA
DELTAI	REFINED ANGLE (RADIANS) DEFINING DECLINATION OF THE LSI POINT (NOMINALLY SET TO DELTAO)
DELTAO	DECLINATION OF THE MOONS POSITION AT TIME TSI
DELT	TIME FROM TSI TO TCA IN SECONDS
DEL	REFINING VALUES FOR ALPHAI, DELTAI, THETAI

DENOM	INTERMEDIATE VARIABLE USED TO LIMIT THE DEL VALUES FOR EACH ITERATION
ECC	DESIRED ECCENTRICITY OF THE LUNAR CONIC
ECEQ	TRANSFORMATION MATRIX FROM ECLIPTIC TO EARTH EQUATORIAL
ECLQ	TRANSFORMATION MATRIX FROM ECLIPTIC TO LUNAR EQUATORIAL
ERR	VECTOR OF DIFFERENCES BETWEEN DESIRED AND NOMINAL VALUES OF B DOT T, B DOT R, ACA
ITAR	LOGIC CONTROLLING INDICATOR =1 IMPROVE ACA ONLY =2 IMPROVE RCA, ICA, ACA
ITER	ITERATION COUNTER FOR NOMINAL TRAJECTORIES
IT	ITERATION COUNTER FOR PERTURBED TRAJECT-ORIES
I	INDEX
J	INDEX
K	INDEX
ONEMAT	UNIT DUMMY MATRIX FOR CALL TO IMPACT
PAI	DUMMY VARIABLE FOR CALL TO LUNCON WHEN ITAR=1
PARP	DUMMY VARIABLE FOR CALL TO LUNCON WHEN ITAR=1
PARTA	INTERMEDIATE VARIABLE USED TO REFINE ACA WHEN ITAR=1
PARTW	INTERMEDIATE VARIABLE USED TO COMPUTE DELT
PARTX	INTERMEDIATE VARIABLE USED TO COMPUTE DELT
PARTY	INTERMEDIATE VARIABLE USED TO COMPUTE DELT
PARTZ	INTERMEDIATE VARIABLE USED TO COMPUTE DELT
PHI	MATRIX RELATING PERTURBATIONS IN ALPHAI, DELTAI, AND THETAI TO CHANGES IN B DOT T, B DOT R, AND ACA

PSI	TARGETING MATRIX RELATING PERTURBATIONS IN B DOT T, B DOT R, AND ACA TO CHANGES IN ALPHAI, DELTAI, AND THETAI
PTAR	PERTURBED BALUES OF AA, BDT, BDR, USED TO CALCULATE PHI
RAD	CONVERTS DEGREES TO RADIANS
RMAG	MAGNITUDE OF THE RMQ VECTOR
SIGMAL	LAUNCH AZIMUTH SET IN LUNCON (NOMINALLY 90 DEGREES)
TAR	NOMINAL VALUES OF AA, BDT AND BDR USED TO CALCULATE PHI
THEI	PERTURBED VALUES OF THETAI USED TO SOLVE FOR RCA, ICA, AND ACA
THETAI	REFINED ANGLE (RADIANS) DEFINING RIGHT ASCENSION OF THE LSI POINT (NOMINALLY SET TO THETAO)
THETAO	RIGHT ASCENSION OF THE MOONS POSITION AT TIME TSI
TM	CONSTANT VALUE OF SECONDS PER DAY
TSICA	DUMMY ARGUMENT FOR CALL TO IMPACT

COMMON COMPUTED/USED:	DTAR	EQLQ	ITAG	RMQ	RSI
	TMU	TSI			
COMMON COMPUTED:	EMU	NO	RME		
COMMON USED:	BCON	DECLIN	FIVE	ONE	OTAR
	PCON	PMASS	RCA	SMA	TCA
	TSPH	TTOL	TWO	XP	ZERO

LUNTAR Analysis

LUNTAR generates a patched conic trajectory arriving at closest approach to ghe Moon at a specified time t_{CA} and meeting prescribed target values at that point as well as standard launch quantities. The target parameters are

t_{CA}	Julian date of required closest approach (CA) referenced 1900
r_{CA}	Radius of CA
i_{CA}	Inclination (relative to lunar equator) at CA[1]
a_{CA}	Semi-major axis at CA

The launch parameters

ϕ_L	Launch site latitude
θ_L	Launch site longitude
Σ_L	Launch azimuth (nominally set to 90°)
r_p	Parking orbit radius

The eccentricity of the moon-centered hyperbola may be computed

$$e_{CA} = 1 - \frac{r_{CA}}{a_{CA}} \tag{1}$$

where $a_{CA} < 0$. The hyperbolic time Δt to go from R_{SI} (radius of lunar sphere of influence (LSI)) to periapsis may be computed from

$$\Delta t = f(\mu_M, a_{CA}, e_{CA}, R_{SI}) \tag{2}$$

where μ_M is the lunar gravitational constant. The time at which the probe should intersect the LSI is then

$$t_{SI} = t_{CA} - \Delta t \tag{3}$$

1 The inclination must be specified according to the format described in IMPACT. For $0 \leq i < 90^\circ$ the inclinations $\pm i$ prescribe posigrade orbits while $180^\circ \pm i$ define retrograde orbits. The positive signs denote approaches from the north, the negative signs designate southern approaches.

The position $\vec{R}_{ME}$ and velocity $\vec{V}_{ME}$ of the moon at t_{SI} relative to the earth in earth ecliptic (EC) coordinates are computed by calling ORB and EPHEM. Transformation matrices ϕ_{ECEQ} and ϕ_{EQLQ} defining transformations from EC to EQ (earth equatorial) and EQ to LQ (lunar equatorial) respectively are then computed by PECEQ. The position and velocity of the moon in the EQ system are

$$\vec{R}_{MQ} = \vec{\phi}_{ECEQ} \vec{R}_{ME}$$
$$\vec{V}_{MQ} = \vec{\phi}_{ECEQ} \vec{V}_{ME} \tag{4}$$

Call the point of intersection of the vector R_{MQ} with the LSI the bullseye point. Then in moon-centered Earth-equatorial coordinates the vector to the bullseye point is given by

$$\vec{r}_B = -\left(\frac{\vec{R}_{MQ}}{R_{MQ}}\right) R_{SI} \tag{5}$$

From this vector one can calculate a set of angular coordinates (δ_o, θ_o) of the bullseye point. Any other point on the LSI is determined by giving general coordinates (δ, θ) = ($\delta_o + \Delta\delta$, $\theta_o + \Delta\theta$).

Now let such a set of coordinates by given. They determine a vector $\vec{R}_I$ from earth to the LSI (in the EQ-system). The vector $\vec{R}_I$ along with the launch parameters ϕ_L, θ_L, Σ_L then determines the plane of the Earth-LSI transfer (see LUNCON). Now let α be measured counter-clockwise in that plane from $-\vec{R}_I$. The parameter α specifies the location of the perigee point of the transfer conic, thus the vector to perigee is fixed as $\vec{r}_P$ where the perigee magnitude r_p is fixed as input. The vectors $\vec{r}_p$ and $\vec{R}_I$ then determine a unique conic for the Earth-LSI phase (see LUNCON). Let the state at the LSI on that conic (relative to Earth-equatorial coordinates) be denoted by $\vec{R}_I, \vec{V}_I$. The state relative to the moon may then be computed as

$$\vec{r}_I = \vec{R}_I - \vec{R}_{MQ}$$
$$\vec{v}_I = \vec{V}_I - \vec{V}_{MQ} \tag{6}$$

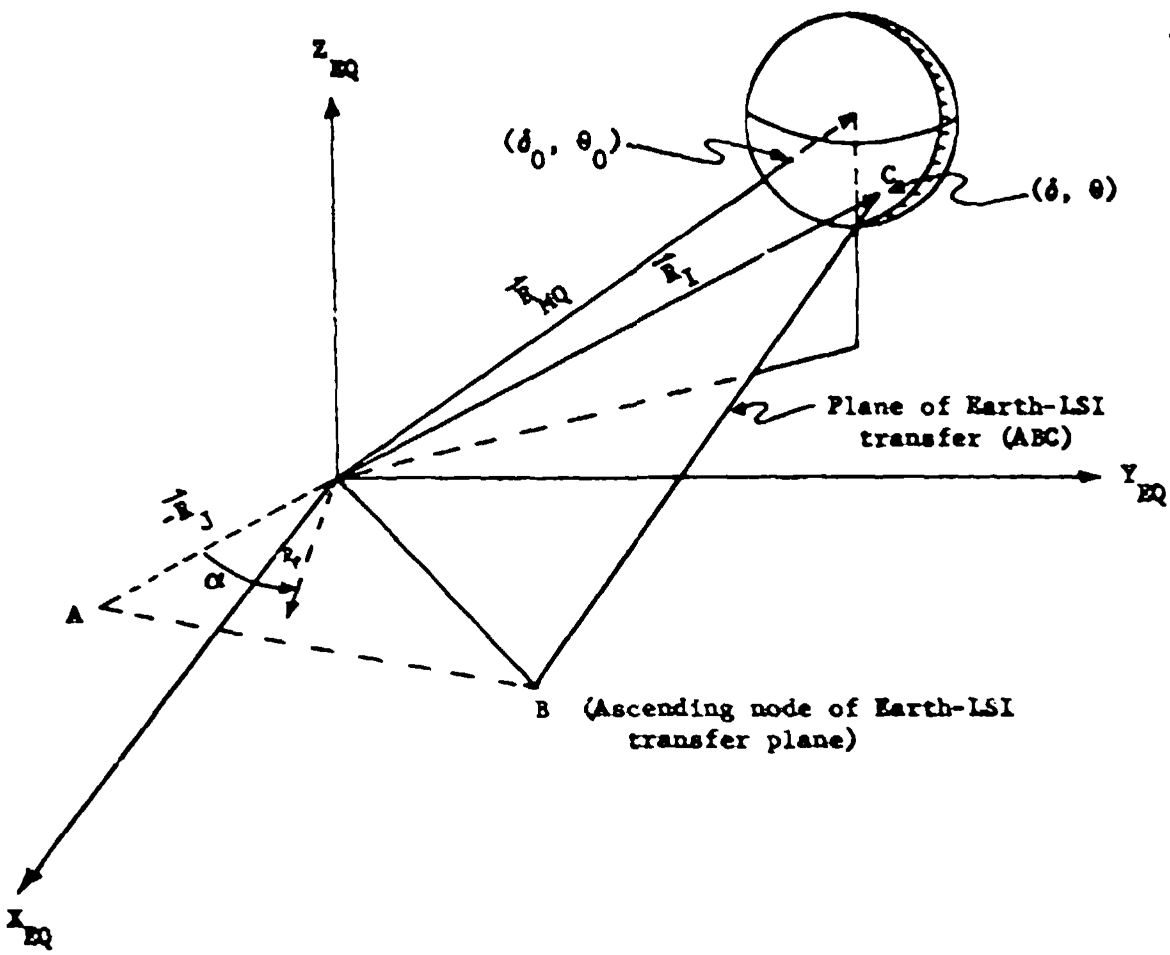


Figure 1. Lunar Patched Conic Targeting

Thus the elements relative to the moon may be computed from standard conic formula. The three angles (δ, θ, α) form a set of independent controls to be varied to meet the three constraints (r_{CA}, i_{CA}, a_{CA}). The controls are depicted in Figure 1.

LUNTAR uses the standard Newton-Raphson algorithm to refine the controls to meet the constraints. This targeting is done in two stages. In the first stage the controls δ and θ are held fixed at the bullseye point (δ_0, θ_0) while α is varied until the semi-major axis target a_{CA} is met. Then all three controls are varied to satisfy the three target constraints. The preliminary targeting of a_{CA} is essential to the success of the procedure. Once the in initial targeting is completed, the semi major axis of future

iterations in the second stage will not vary much from the target value a_{CA}. For such iterates the excess hyperbolic velocity at the moon will be generally constant. This permits the substitution of the auxiliary impact plane parameters B·T and B·R for the less linear parameters of r_{CA} and i_{CA} (see IMPACT). In LUNTAR the impact plane parameters are referenced to the LQ system.

The procedure may now be described in detail. Suppose that in the first stage of targeting the current value of α is α_K. Using the controls $(\alpha_K, \delta_0, \theta_0)$ the resulting semi-major axis is found to be a_K (LUNCON). A perturbed value for the first control is then used $(\alpha_K + \Delta\alpha, \delta_0, \theta_0)$ producing a perturbed value of semi-major axis $(a_K + \Delta a)$. The (k+1)st value of α is then given by the standard numerical differencing approximation

$$\alpha_{K+1} = \alpha_K + \frac{\Delta\alpha}{\Delta a}(a_{CA} - a_K) \qquad (7)$$

The second stage of the targeting of the lunar patched conic uses the vector analogue of the above procedure. The current iterate $(\alpha_K, \delta_K, \theta_K)$ is input to LUNCON to obtain the current target values $(a_K, B{\cdot}T_K, B{\cdot}R_K)$. The target values B·T and B·R are determined from subroutine IMPACT and the errors of the kth iterate are computed (e_a, e_{BT}, e_{BR}). If all three errors are within tolerances, the procedure is terminated. Otherwise the sensitivity matrix ϕ is computed by numerical differencing as in the first stage

$$\phi = \begin{bmatrix} \frac{\Delta a_\alpha}{\Delta\alpha} & \frac{\Delta a_\delta}{\Delta\delta} & \frac{\Delta a_\theta}{\Delta\theta} \\ \frac{\Delta B{\cdot}T_\alpha}{\Delta\alpha} & \frac{\Delta B{\cdot}T_\delta}{\Delta\delta} & \frac{\Delta B{\cdot}T_\theta}{\Delta\theta} \\ \frac{\Delta B{\cdot}R_\alpha}{\Delta\alpha} & \frac{\Delta B{\cdot}R_\delta}{\Delta\delta} & \frac{\Delta B{\cdot}R_\theta}{\Delta\theta} \end{bmatrix} \qquad (8)$$

The inverse of ϕ is the targeting matrix. The k+1 iterate is then defined to be

$$\begin{bmatrix} \alpha \\ \delta \\ \theta \end{bmatrix}_{K+1} = \begin{bmatrix} \alpha \\ \delta \\ \theta \end{bmatrix}_{K} + \beta^{-1} \begin{bmatrix} a_{CA} - a_K \\ B\cdot T - B\cdot T_K \\ B\cdot R - B\cdot R_K \end{bmatrix} \qquad (9)$$

This procedure is repeated until convergence is achieved.

PROGRAM MAIN

PURPOSE: TO CONTROL THE SIMULATION OVERLAY SCHEME

SUBROUTINES SUPPORTED: NONE

SUBROUTINES REQUIRED: DATAS SIMUL PRNTS4

LOCAL SYMBOLS: IRUNX TOTAL NUMBER OF DATA CASES

IRUN DATA CASE COUNTER

SUBROUTINE MATIN

PURPOSE: TO COMPUTE THE INVERSE OF A MATRIX.

CALLING SEQUENCE: CALL MATIN(A,R,N)

ARGUMENTS	A(N,N)	I	MATRIX TO BE INVERTED
	R(N,N)	O	RESULTANT INVERSE OF MATRIX A
	N	I	DIMENSION OF A AND R

SUBROUTINES SUPPORTED: HYELS NAVM BIAIM POICOM GUISS TARMAX GUID LUNTAR MULTAR

SUBROUTINES REQUIRED: NONE

LOCAL SYMBOLS:	AL	A(LL) + S (INTERMEDIATE VARIABLE)
	ALBAR	INTERMEDIATE VARIABLE
	B	INTERMEDIATE VECTOR
	DETR	INTERMEDIATE VECTOR
	G	INTERMEDIATE VECTOR
	IX	INTERMEDIATE VECTOR
	KR	DIMENSION OF A
	MIXI	INTERMEDIATE VARIABLE
	MIXJ	INTERMEDIATE VARIABLE
	MIXL	INTERMEDIATE VARIABLE
	S	INTERMEDIATE VARIABLE
	X	INTERMEDIATE VARIABLE
	XOFF	INTERMEDIATE VARIABLE

COMMON USED: EM7 EM9 ONE ZERO

SUBROUTINE MENO

PURPOSE: COMPUTE ASSUMED MEASUREMENT NOISE COVARIANCE MATRIX IN THE ERROR ANALYSIS PROGRAM

CALLING SEQUENCE: CALL MENO(MMCODE,ICODE)

ARGUMENT:	ICODE	I	INTERNAL CODE USED TO DISTINGUISH BETWEEN THE TWO ALTERNATIVES LISTED ABOVE
	MMCODE	I	MEASUREMENT MODEL CODE

SUBROUTINES SUPPORTED: ERRANN

COMMON COMPUTED: R

COMMON USED: IMNF MNCN

MENØ Analysis

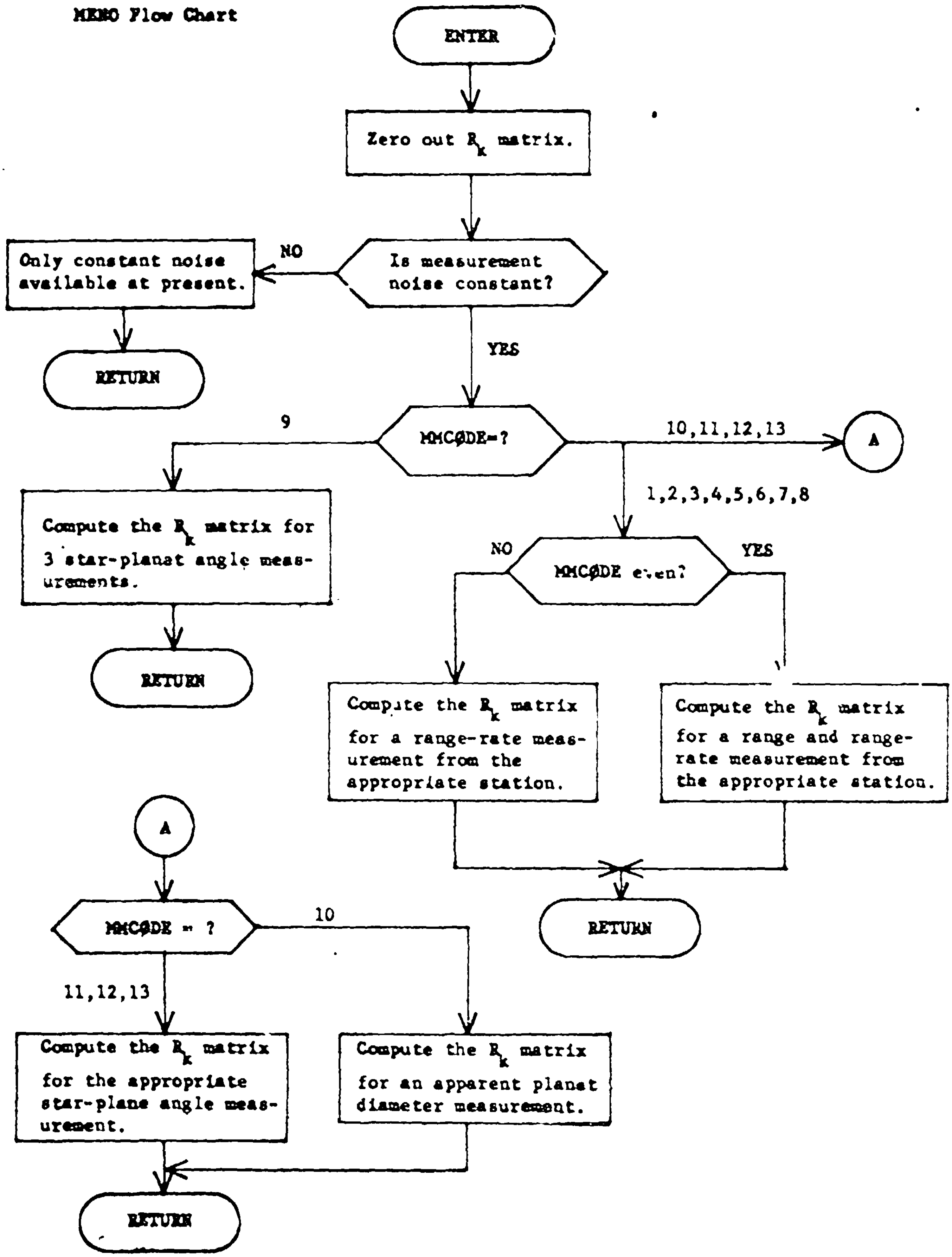
The linearized observation equation employed by the navigation process is given by

$$\delta Y_k = H_k^A \, \delta X_k^A + \eta_k$$

where δY_k is the measurement deviation from the nominal measurement, H_k^A is the augmented observation matrix, δX_k^A is the augmented state deviation from the nominal augmented state, and η_k is the assumed measurement noise.

The function of subroutine MENØ is to compute the measurement noise covariance matrix R_k which describes the statistics of η_k. The constant variances for the measurement noises associated with all available measurement devices are stored in the vector MNCN. Subroutine MENØ selects the appropriate elements from this vector to construct the measurement noise covariance matrix R_k.

MENO Flow Chart



SUBROUTINE MENOS

PURPOSE: COMPUTE ASSUMED AND ACTUAL MEASUREMENT NOISE COVARIANCE MATRICES IN THE SIMULATION PROGRAM

CALLING SEQUENCE: CALL MENOS(MMCODE,ICODE)

ARGUMENT:	ICODE	I	INTERNAL CODE USED TO DISTINGUISH BETWEEN THE TWO ALTERNATIVES LISTED ABOVE
	MMCODE	I	MEASUREMENT MODEL CODE

SUBROUTINES SUPPORTED: SIMULL

COMMON COMPUTED/USED:	R				
COMMON COMPUTED:	AR				
COMMON USED:	AVARM	IAMNF	IMNF	MNCN	ZERO

MENØS Analysis

The linearized observation equation employed by the navigation process is given by

$$\delta Y_k = H_k^A \, \delta X_k^A + \eta_k$$

where δY_k is the measurement deviation from the nominal measurement, H_k^A is the augmented observation matrix, δX_k^A is the augmented state deviation from the nominal augmented state, and η_k is the assumed measurement noise. The actual measurement Y_k^a is given by

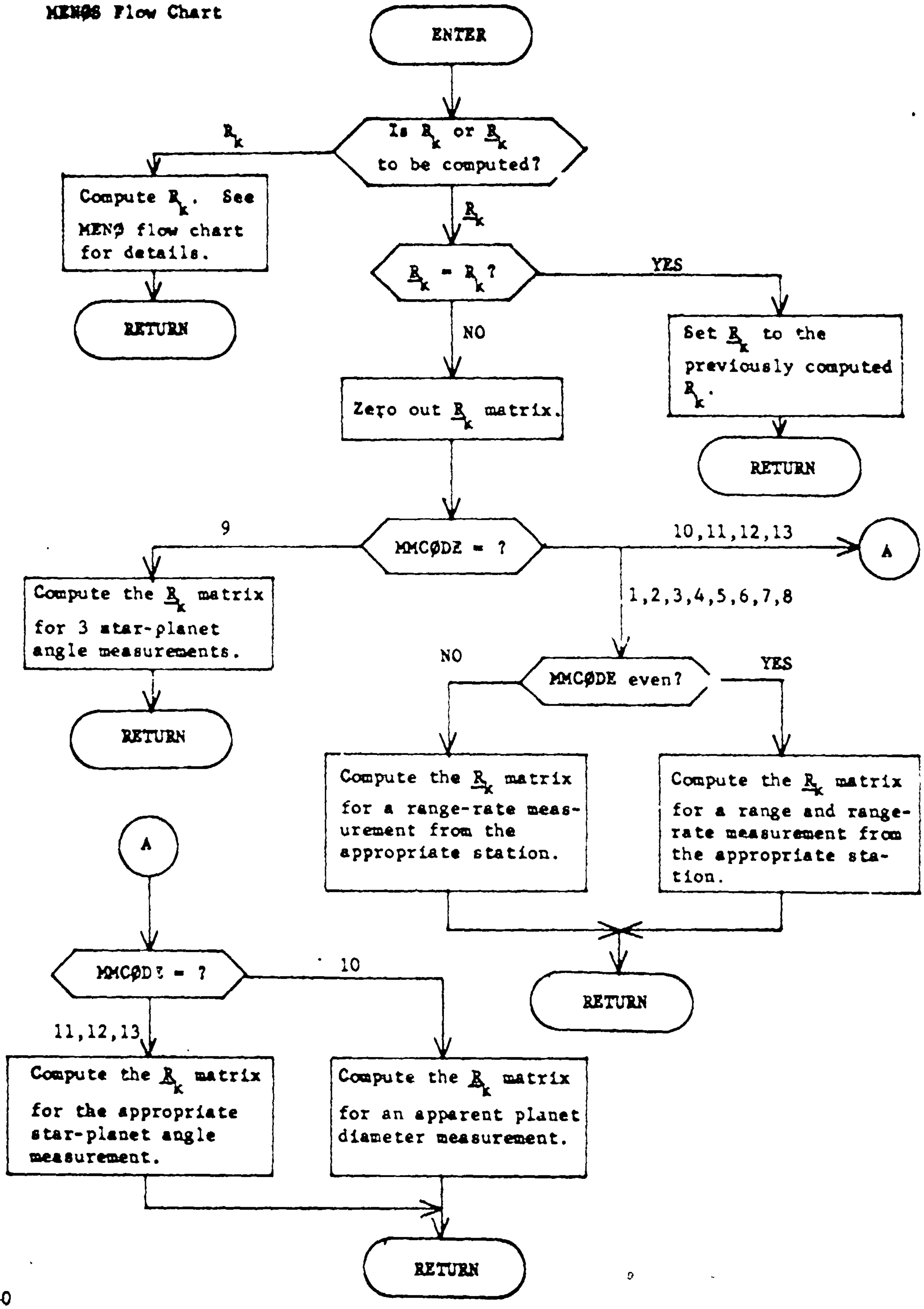
$$Y_k^a = \underline{Y}_k + b_k + \nu_k$$

where $\underline{Y}_k$ is the ideal measurement, which would be made in the absence of instrumentation errors, b_k is the actual measurement bias, and ν_k represents the actual measurement noise.

Subroutine MENOS performs two functions. It's first function, which is identical to that of subroutine MENO, is to compute the measurement noise covariance matrix R_k which describes the statistics of noise η_k. The constant variances for the assumed measurement noises associated with all available measurement devices are stored in the vector MNCN. Subroutine MENOS selects the appropriate elements from this vector to construct the measurement noise covariance matrix R_k.

The second function of MENOS is to compute the measurement noise covariance matrix $\underline{R}_k$ which describes the statistics of the actual noise ν_k. The constant variances for the actual measurement noises associated with all available measurement devices are stored in the vector AVARM. Subroutine AVARM selects the appropriate elements from this vector to construct the measurement noise covariance matrix $\underline{R}_k$.

MENØS Flow Chart



SUBROUTINE MULCON

PURPOSE: TO PROPAGATE A SET OF CARTESIAN COORDINATES ALONG A LUNAR MULTI-CONIC TRAJECTORY OVER A SPECIFIED TIME INTERVAL.

CALLING SEQUENCE: CALL MULCON(SEI,TLI,TF,DT,SMF)

ARGUMENTS:

ARGUMENTS:	SEI(6)	I	INITIAL SPACECRAFT GEOCENTRIC STATE
	TLI	I	INITIAL INJECTION TIME (JD EPOCH 1950)
	TF	I	TIME INTERVAL OF PROPAGATION
	DT	I	STEP SIZE USED IN MULTICONIC PROPAGATION
	SMF(6)	O	FINAL SPACECRAFT SELENOCENTRIC STATE

SUBROUTINES SUPPORTED: MULTAR

SUBROUTINES REQUIRED: CPROP EPHEM ORB

LOCAL SYMBOLS:

LOCAL SYMBOLS:	ALNGTH	CONVERTS KILOMETERS TO ASTRONOMICAL UNITS
	A	PERTURBING ACCELERATION VECTOR OF THE MOON OVER THE ITERATION INTERVAL
	COSF	COSINE OF TRUE ANOMALY ON SELONOCENTRIC CONIC AT END OF ITERATION INTERVAL
	DF	FINAL TIME USED IN ITERATION INTERVAL
	DI	INITIAL TIME USED IN ITERATION INTERVAL
	EMM	MAGNITUDE OF FIRST THREE ELEMENTS OF EM
	EM	GEOCENTRIC ECLIPTIC STATE OF MOON
	IDONE	STOPPING CONDITION INDICATOR =0 PROPAGATION CONTINUES =1 STOPPING CONDITION REACHED
	I	INDEX
	TIM	JULIAN DATE OF FINAL TIME ON THE ITERATION INTERVAL
	TM	CONVERTS SECONDS TO DAYS
	W	SPACECRAFT VELOCITY VECTOR WITH RESPECT TO EARTH AND/OR MOON BEFORE AND AFTER LUNAR PERTURBATIONS AT DI AND DF

XM	MAGNITUDE OF THE X VECTOR
X	GEOCENTRIC POSITION OF SPACECRAFT AT DI AND GEOCENTRIC POSITION OF MOON AT DF
Y	GEOCENTRIC VELOCITY OF SPACECRAFT AT DI AND GEOCENTRIC VELOCITY OF MOON AT DF
Z	SPACECRAFT POSITION VECTOR WITH RESPECT TO EARTH AND/OR MOON AT DI AND DF BEFORE AND AFTER LUNAR PERTURBATIONS

COMMON COMPUTED: NO

COMMON USED: EMU TMU TMO XP ZERO

MULCØN Analysis

The equations of motion of a spacecraft traveling under the influence of the earth and moon may be written

$$\ddot{\vec{r}}_E = -\frac{\mu_E \vec{r}_E}{r_E^3} - \frac{\mu_M \vec{r}_M}{r_M^3} - \frac{\mu_M \vec{R}_{EM}}{r_{EM}^3} \tag{1}$$

where $\vec{r}_E$, $\vec{r}_M$, $\vec{R}_{EM}$ are the position vectors of the spacecraft-to-earth, the spacecraft-to-moon, and the moon-to-earth respectively and μ_E, μ_M are the gravitational constants of the earth and moon respectively.

The multi-conic approximation of the solution to (1) proceeds as follows. Let $\vec{r}_{E,k}$, $\vec{V}_{E,k}$ be the geocentric state at some time t_k. This state is propagated by conic formulae to obtain an estimate of the geocentric state at time $t_{k+1} = t_k + \Delta t$ given by $\vec{r}\,'_{E,k+1}$, $\vec{V}\,'_{E,k+1}$.

To account for the third term perturbations, the state of the moon relative to the earth at the two timepoints is computed, denoted by $(\vec{R}_{EM,k}, \vec{V}_{EM,k})$ and $(\vec{R}_{EM,k+1}, \vec{V}_{EM,k+1})$. The average value of this acceleration is then determined from

$$\vec{A} = -\frac{\mu_M}{2}\left[\frac{\vec{R}_{EM,k}}{R_{EM,k}^3} + \frac{\vec{R}_{EM,k+1}}{R_{EM,k+1}^3}\right] \tag{2}$$

The corrected geocentric state is then given by

$$\begin{aligned} \vec{r}\,''_{E,k+1} &= \vec{r}\,'_{E,k+1} + \frac{1}{2}\vec{A}\,(\Delta t)^2 \\ \vec{V}\,''_{E,k+1} &= \vec{V}\,'_{E,k+1} + \vec{A}\,\Delta t \end{aligned} \tag{3}$$

The effect of the direct lunar perturbations is then added. The state of the spacecraft relative to the moon is first computed

$$\begin{aligned} \vec{r}\,'_{M,k+1} &= \vec{r}\,''_{E,k+1} - \vec{R}_{EM,k+1} \\ \vec{V}\,'_{M,k+1} &= \vec{V}\,''_{E,k+1} - \vec{V}_{EM,k+1} \end{aligned} \tag{4}$$

This state is then propagated linearly backwards in time over the time interval Δt to obtain

$$\vec{r}'_{M,k} = \vec{r}'_{M,k+1} - \vec{v}_{M,k+1}\,\Delta t \qquad (5)$$

$$\vec{v}'_{M,k} = \vec{v}'_{M,k+1}$$

This state is now propagated forward in a selenocentric conic to obtain a final state relative to the moon $(\vec{r}_{M,k+1}, \vec{v}_{M,k+1})$. The geocentric state of the spacecraft at time t_{k+1} after considering all terms of (1) is then given by

$$\vec{r}_{E,k+1} = \vec{r}_{M,k+1} + \vec{R}_{EM,k+1}$$

$$\vec{v}_{E,k+1} = \vec{v}_{M,k+1} + \vec{V}_{EM,k+1} \qquad (6)$$

This completes one cycle of the multi-conic propagation.

The multi-conic propagation proceeds until an input final time is reached or until the selenocentric conic passes through pericynthion.

Reference: Byrnes, D. V. and Hooper, H. L., Multi-Conic: A Fast and Accurate Method of Computing Space Flight Trajectories, AAS/AIAA Astrodynamics Conference, Santa Barbara, Cal., 1970, AIAA Paper 70-1062.

SUBROUTINE MULTAR

PURPOSE: TO CALCULATE THE TRANSLUNAR INJECTION CONDITIONS FROM TARGETED PATCHED-CONIC CONDITIONS AND CALLS VMP TO PERFORM THE NOMINAL TRAJECTORY NEEDED BY ITERAT.

CALLING SEQUENCE: CALL MULTAR

SUBROUTINES SUPPORTED: LUNA

SUBROUTINES REQUIRED: MULCON CAREL ELCAR EPHEM IMPACT MATIN ORB PECEQ TIME

LOCAL SYMBOLS:

AE	SEMI-MAJOR AXIS OF THE EARTH-ECLIPTIC, TARGETED PATCHED-CONIC TRAJECTORY
ATARN	NOMINAL VALUES OF THE TARGET VARIABLES
ATAR	DESIRED VALUES OF THE TARGET VARIABLES
ATOL	TOLERANCES OF TARGET VARIABLES
BCOR	MAXIMUM STEPS ALLOWED IN ITERATIVE CORRECTION OF CONTROL VARIABLES
BJ	ZERO TRUE ANOMALY USED TO DEFINE PERIGEE OF THE TARGETED PATCHED-CONIC TRAJECTORY
BSTEP	MULTI-CONIC STEP SIZE (HOURS)
CHI	SENSITIVITY MATRIX RELATING PERTURBATIONS IN CONTROL VARIABLES TO CHANGES IN TARGET VARIABLES
DELP	VALUE USED TO PERTURB TLI FOR CONSTRUCTION OF CHI
DELTN	NOMINAL TIME FOR PROPAGATION
DELT	INTEGRATION TIME TO BE USED, AND TIME ACTUALLY USED, IN THE MULTI-CONIC PROPAGATION
DELV	VALUE USED TO PERTURB VELOCITY COMPONENTS OF RT FOR CONSTRUCTION OF CHI
DV	CORRECTION ACTUALLY ADDED TO CONTROL VARIABLES
ECEQP	TRANSFORMATION MATRIX FROM EARTH ECLIPTIC TO LUNAR EQUATORIAL
EECEQ	TRANSFORMATION MATRIX FROM EARTH ECLIPTIC

	TO EARTH EQUATORIAL
EE	ECCENTRICITY OF THE EARTH-ECLIPTIC, TARGETED PATCHED-CONIC TRAJECTORY
ERR	ITERATE ERRORS IN TARGET CONDITIONS
FAC	INTERMEDIATE VARIABLE USED TO CHECK FOR MAXIMUM STEP
FMAG	INTERMEDIATE VARIABLE USED TO COMPUTE PHIA, PHIB, PHIC
HYT	HYPERBOLIC TIME TO LUNAR PERIAPSIS (DAYS)
IDATE	CALENDAR DATE OF INJECTION
IST	INDICATOR FOR CONTROL VARIABLE PERTURBATION
IT	ITERATIONS COUNTER
I	INDEX
JERTH	INDEX OF EARTH IN F-ARRAY
JMOON	INDEX OF MOON IN F-ARRAY
J	INDEX
K	INDEX
MITS	MAXIMUM NUMBER OF ITERATIONS ALLOWED
NDEX	SAME AS JERTH
PERMN	MINIMUM PERTURBATION OF CONTROL VARIABLES FOR CONSTRUCTION OF CHI
PERMX	MAXIMUM PERTURBATION OF CONTROL VARIABLES FOR CONSTRUCTION OF CHI
PERT	PERTURBATION VALUES USED TO CONSTRUCT CHI
PHIA	TRANSFORMATION MATRIX FROM RTN TO EC SYSTEM AT TLI
PHIB	TRANSFORMATION MATRIX FROM RTN TO EC SYSTEM AT PERTURBED TLI
PHIC	PRODUCT OF PHIB AND TRANSPOSE OF PHIA
PPE	DUMMY VARIABLE FOR CALL TO CAREL

PSI	TARGET MATRIX (INVERSE OF CHI) RELATING PERTURBATIONS IN TARGET VARIABLES TO CHANGES IN CONTROL VARIABLES
PTAR	PERTURBED TARGET VALUES
PV	PREDICTED CORRECTIONS TO CONTROL VARIABLES
QQE	DUMMY VARIABLE FOR CALL TO CAREL
REMAG	DUMMY VARIABLE FOR CALL TO ELCAR
REPET	MINIMUM ALLOWABLE INJECTION TIME DIFFERENCE IN KTH AND K+2 ITERATIONS TO AVOID REPETITION-TRAP CORRECTION
RM	MAGNITUDE OF THE SCM POSITION VECTOR
RSE	INJECTION STATE IN EARTH EQUATORIAL SYSTEM AT TLI
RS	ROTATED INJECTION STATE FOR TIME DIFFERENTIAL
RT	INJECTION STATE USED IN PERTURBED MULIT-CONIC PROPAGATIONS
SCM	FINAL STATE IN LUNAR ECLIPTIC SYSTEM ON THE MULTI-CONIC
SEC	SECONDS OF CALENDAR DATE OF TLI
STEP	MULTI-CONIC STEP SIZE (SECONDS)
STLI	ORIGINAL VALUE OF TLI, RESTORED FOR SUCESSIVE ITERATIONS
TAE	TRUE ANOMALY OF EARTH-ECLIPTIC TARGETED PATCHED-CONIC TRAJECTORY
TBR	DUMMY VARIABLE FOR CALL TO IMPACT
TBT	DUMMY VARIABLE FOR CALL TO IMPACT
TFP	TIME OF FLIGHT FROM PERIGEE OF THE EARTH-ECLIPTIC, TARGETED PATCHED-CONIC TRAJECTORY
TIMM1	INJECTION DATE ON K-1 ITERATION
TIMM2	INJECTION DATE ON K-2 ITERATION

TLI	INJECTION JULIAN DATE
TTP	DUMMY VARIABLE FOR CALL TO ELCAR
VEMAG	DUMMY VARIABLE FOR CALL TO ELCAR
VX	MAGNITUDE OF THE SCM VELOCITY VECTOR
WE	ARGUMENT OF PERIAPSIS OF THE EARTH-ECLIPTIC, TARGETED PATCHED-CONIC TRAJECTORY
WWE	DUMMY ARGUMENT FOR CALL TO CAREL
XIE	INCLINATION OF THE EARTH-ECLIPTIC, TARGETED PATCHED-CONIC TRAJECTORY
XNE	LONGITUDE OF ASCENDING NODE OF THE EARTH-ECLIPTIC, TARGETED PATCHED-CONIC TRAJECTORY

COMMON COMPUTED/USED:	NO	RI			
COMMON COMPUTED:	ICOORD	RIN	TIN		
COMMON USED:	ALNGTH	CAI	EMU	F	HALF
	KUR	NBOD	NB	ONE	RCC
	SMA	TAR	TCA	TEN	TMU
	TM	TSI	TWO	ZERO	

MULTAR Analysis

Let the earth equatorial state of the probe at the LSI as computed from the patched conic targeting be denoted $\vec{r}_{LS}$, $\vec{v}_{LS}$. Subroutine CAREL is called to compute the conic elements and conic time from perigee Δt based on the geocentric conic. The time of injection is then computed as

$$t_{TLI} = t_{SI} - \Delta t \tag{1}$$

The position and velocity of the probe at t_{TLI} is given by the state along the conic at perigee (true anomaly of zero) and determined by ELCAR to be $\vec{r}_{TLI}$, $\vec{v}_{TLI}$. If Φ_{ECEQ} is the transformation matrix from the EC (earth ecliptic) to the EQ (earth equatorial) system, then the patched conic injection state in EC coordinates is

$$\begin{aligned} \vec{r}_I &= \Phi^T_{ECEQ}\, \vec{r}_{TLI} \\ \vec{v}_I &= \Phi^T_{ECEQ}\, \vec{v}_{TLI} \end{aligned} \tag{2}$$

Since the earth is revolving about the E-M barycenter in time, the EC injection state must be rotated if an earlier or later injection time is to be used. The necessary rotation matrix may be easily computed through the introduction of the R-T-W coordinate system. Let the state of the earth at some time t_k in BC (barycentric ecliptic) coordinates be denoted $\vec{R}_k$, $\vec{V}_k$. Construct the $\hat{R}$-$\hat{T}$-$\hat{W}$ system at that point as

$$\hat{R}_k = \frac{\vec{R}_k}{R_k} \qquad \hat{W}_k = \frac{\vec{R}_k \times \vec{V}_k}{\left|\vec{R}_k \times \vec{V}_k\right|} \qquad \hat{T}_k = \hat{W}_k \times \hat{R}_k \tag{3}$$

The transformation matrix from the $\hat{R}_k$-$\hat{T}_k$-$\hat{W}_k$ system to the ecliptic system to the ecliptic system is then given by

$$\Phi_k = \left[\hat{R}_k \;\middle|\; \hat{T}_k \;\middle|\; \hat{W}_k \right] \tag{4}$$

At a time t_{k+1} the state of the earth in BC coordinates is given by $\vec{R}_{k+1}$, $\vec{V}_{k+1}$ and the transformation from the $\hat{R}_{k+1}$ - $\hat{T}_{k+1}$ - $\hat{W}_{k+1}$ system to ecliptic coordinates is given by Φ_{k+1} in accordance with (4). Injection

states at times t_k and t_{k+1} will be called "equivalent" if they are identical when expressed in the pertinent $\hat{R}$-$\hat{T}$-$\hat{W}$ system. Therefore if $(\vec{r}_k, \vec{v}_k)$ is the injection state in EC coordinates at time t_k, the equivalent state in EC coordinates at t_{k+1} is given by

$$\begin{bmatrix} \vec{r}_{k+1} \\ \vec{v}_{k+1} \end{bmatrix} = \begin{bmatrix} \Psi_{k+1,k} & 0 \\ 0 & \Psi_{k+1,k} \end{bmatrix} \begin{bmatrix} \vec{r}_k \\ \vec{v}_k \end{bmatrix} \tag{5}$$

where the rotation matrix Ψ is defined by

$$\Psi_{k+1,k} = \Phi_{k+1} \Phi_k^T \tag{6}$$

The targeting algorithm used by MULTAR may now be described. Let the injection state in EC coordinates on the k-th iteration be denoted $(t_k, \vec{r}_k, \vec{v}_k)$. This state is propagated forward using the multi-conic propagator MULCØN to determine a final state $\vec{r}_M, \vec{v}_M$ to the moon in ecliptic coordinates. IMPACT is then called to compute the $B \cdot T_k$, $B \cdot R_k$ and $t_{CA,k}$ actually achieved on the trajectory and the target values of $B^* \cdot T_k$, $B^* \cdot R_k$ required to satisfy the i_{CA} and r_{CA} constraints. The semi-major axis a_k of the k-th iterate is computed from the conic formula

$$a = r_M \left(2 - \frac{r_M v_M^2}{\mu_M} \right)^{-1} \tag{7}$$

Errors in the four target conditions

$$\Delta\tau = \begin{bmatrix} \Delta a \\ \Delta B \cdot T \\ \Delta B \cdot R \\ \Delta t_{CA} \end{bmatrix} = \begin{bmatrix} a - a^* \\ B \cdot T_k - B^* \cdot T_k \\ B \cdot R_k \quad B^* \cdot R_k \\ t_{CA,k} - t_{CA}^* \end{bmatrix} \tag{8}$$

If the error in each parameter is less than the allowable tolerance, the process stops.

If convergence has not been achieved a Newton-Raphson iteration is entered. The four controls are $\vec{v}_{k_x}$, $\vec{v}_{k_y}$, $\vec{v}_{k_z}$, and t_k. For the velocity components a perturbation Δv is added to the pertinent component while the rest of the injection state is held constant before propagating with the multi-conic. For the time perturbation, the rotation matrix Ψ_Δ corresponding to the perturbed time $t_k + \Delta t$ (6) is first computed. The injection state used in the perturbed propagation for time is then $\left[t_k + \Delta t, \Psi_\Delta \vec{r}_k, \Psi_\Delta \vec{v}_k\right]$. A sensitivity matrix is computed using the results of the numerical differencing:

$$X = \begin{bmatrix} \dfrac{\Delta a_x}{\Delta v_x} & \dfrac{\Delta a_y}{\Delta v_y} & \cdots & \\ \dfrac{\Delta B \cdot T_x}{\Delta v_x} & \vdots & & \\ \dfrac{\Delta B \cdot R_x}{\Delta v_x} & \vdots & & \\ \dfrac{\Delta t_{CA_x}}{\Delta v_x} & & & \dfrac{\Delta t_{CA_t}}{\Delta t} \end{bmatrix} \tag{9}$$

where in the term $\dfrac{\Delta\alpha_\beta}{\Delta\beta}$, $\Delta\alpha_\beta$ is the change in the α target parameter produced by the variation of the β control component and $\Delta\beta$ is the change in the β control component. The k+1 iterate controls are then given by

$$\Delta C = \begin{bmatrix} \delta v_x \\ \delta v_y \\ \delta v_z \\ \delta t \end{bmatrix} = X^{-1} \Delta \tau \tag{10}$$

The k+1 injection state is then computed by first determining the injection state after rotation due to the change in injection time and then adding the injection velocity corrections

$$\begin{aligned} t_{k+1} &= t_k + \delta t \\ \vec{r}_{k+1} &= \Psi_\delta \vec{r}_k \\ \vec{v}_{k+1} &= \Psi_\delta \vec{v}_k + \delta\vec{v} \end{aligned} \tag{11}$$

The iteration process is repeated until tolerable errors are met. The converged injection state is then integrated in the virtual mass trajectory.

SUBROUTINE MUND

PURPOSE: TO COMPUTE THE AUGMENTED PORTION OF THE STATE TRANSITION MATRIX WHEN THE GRAVITATIONAL CONSTANT OF THE SUN OR OF THE TARGET PLANET HAS BEEN AUGMENTED TO THE BASIC STATE VECTOR.

CALLING SEQUENCE: CALL MUND(RI,RF,POSS)

ARGUMENT:	RF	I	POSITION AND VELOCITY OF THE VEHICLE AT THE END OF THE TIME INTERVAL
	RI	I	POSITION AND VELOCITY OF THE VEHICLE AT THE BEGINNING OF THE TIME INTERVAL
	POSS	I	DISTANCE OF THE VEHICLE FROM THE TARGET PLANET AT THE INITIAL TIME

SUBROUTINES SUPPORTED: PSIM

SUBROUTINES REQUIRED: NTM

LOCAL SYMBOLS:	IC	COUNTER FOR VARIABLES AUGMENTED TO STATE VECTOR
	IPR	TEMPORARY STORAGE FOR IPRINT
	RPER	ALTERED POSITION AND VELOCITY OF VEHICLE AT FINAL TIME
	SAVE	TEMPORARY STORAGE LOCATION FOR GRAVITATIONAL CONSTANTS OF SUN AND TARGET PLANET

COMMON COMPUTED/USED:	IPRINT	PMASS			
COMMON COMPUTED:	TXU	TXXS			
COMMON USED:	ALNGTH	DELMUP	DELMUS	IAUGDC	IAUGIN
	IAUG	NTMC	NTP	SPHERE	TM

MUND Analysis

The nonlinear equations of motion of the spacecraft can be written symbolically as

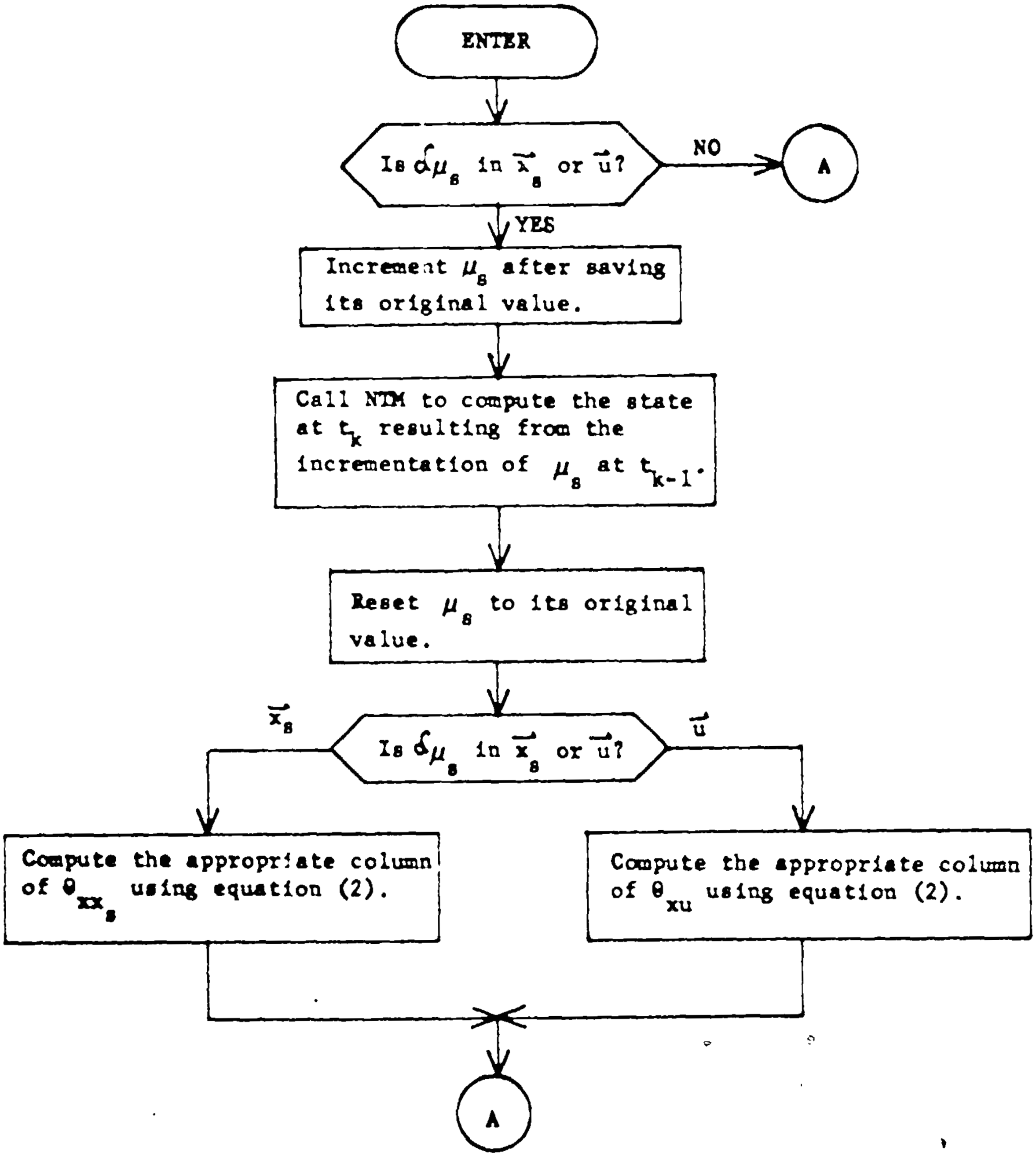
$$\dot{\vec{x}} = \vec{f}\,(\vec{x}, \vec{\mu}, t) \tag{1}$$

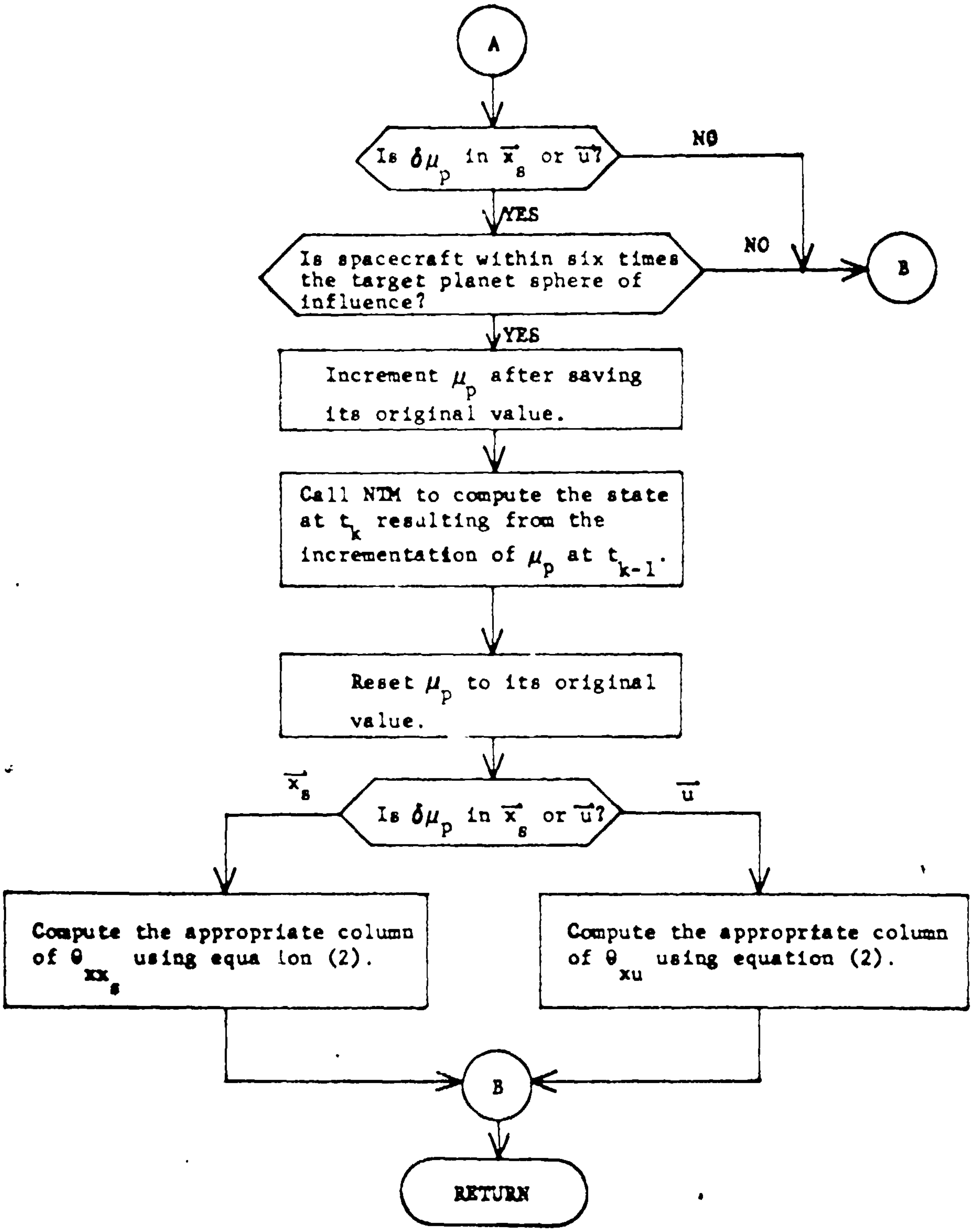
where $\vec{x}$ is the spacecraft position/velocity state and $\vec{\mu}$ is a vector composed of the gravitational constants of the Sun and the target planet.

Suppose we wish to use numerical differencing to compute those columns of θ_{xx_s} and θ_{xu} associated with gravitational constant biases included in the augmented state vector over the time interval $[t_{k-1}, t_k]$. Let $\vec{\theta}_j\,(t_k, t_{k-1})$ represent the column associated with the j-th gravitational constant bias. We assume we have available the nominal states $\vec{x}^*(t_{k-1})$ and $\vec{x}^*(t_k)$, which, of course, were obtained by numerically solving equation (1) using nominal $\vec{\mu}$. To obtain $\vec{\theta}_j(t_k, t_{k-1})$ we increment the j-th gravitational constant bias by the pertinent numerical differencing factor $\Delta\mu_j$ and numerically integrate equation (1) over the interval $[t_{k-1}, t_k]$ to obtain the new spacecraft state $\vec{x}_j(t_k)$, where the j-subscript on the spacecraft state indicates that it was obtained by incrementing the j-th gravitational constant bias. Then

$$\vec{\theta}_j\,(t_k, t_{k-1}) = \frac{\vec{x}_j\,(t_k) - \vec{x}^*(t_k)}{\Delta\mu_j}\,. \tag{2}$$

MUND Flow Chart




A
Is $\delta\mu_p$ in $\vec{x}_s$ or $\vec{u}$?
NO
YES
Is spacecraft within six times the target planet sphere of influence?
NO
B
YES
Increment μ_p after saving its original value.
Call NTM to compute the state at t_k resulting from the incrementation of μ_p at t_{k-1}.
Reset μ_p to its original value.
$\vec{x}_s$
Is $\delta\mu_p$ in $\vec{x}_s$ or $\vec{u}$?
$\vec{u}$
Compute the appropriate column of Θ_{xx_s} using equation (2).
Compute the appropriate column of Θ_{xu} using equation (2).
B
RETURN

SUBROUTINE NAVM

PURPOSE: TO PROPAGATE COVARIANCE MATRIX PARTITIONS P, CXXS, CXU, CXV, PS, CXSU, CXSV FROM THE TIME OF THE LAST MEASUREMENT OR EVENT TO THE PRESENT TIME USING A CONSIDER RECURSIVE ALGORITHM.

CALLING SEQUENCE: CALL NAVM(NR,ICODE)

ARGUMENT: ICODE I INTERNAL CODE WHICH DETERMINES IF A MEASUREMENT IS BEING PROCESSED

NR I NUMBER OF ROWS IN THE OBSERVATION MATRIX

SUBROUTINES SUPPORTED: SIMULL SETEVS GUISIM PRESIM ERRANN SETEVN GUIDM PRED

SUBROUTINES REQUIRED: MATIN

LOCAL SYMBOLS

AJ	MEASUREMENT RESIDUAL COVARIANCE MATRIX AND ITS INVERSE
AKW	INTERMEDIATE ARRAY
DUM	INTERMEDIATE VECTOR
PSAVE	INTERMEDIATE ARRAY
SUM	INTERMEDIATE VARIABLE
SW	INTERMEDIATE ARRAY

COMMON COMPUTED/USED:

AK	CXSUP	CXSU	CXSVP	CXSV
CXUP	CXU	CXVP	CXV	CXXSP
CXXS	PP	PSP	PS	P
S				

COMMON USED:

AL	AM	G	H	NDIM1
NDIM2	NDIM3	ONE	PHI	Q
R	TXU	TXXS	U0	V0
ZERO				

MAVM Analysis

The augmented deviation state vector is defined as

$$\vec{x}^A = \left[\vec{x},\ \vec{x}_s,\ \vec{u},\ \vec{v}\right]^T$$

where

$\vec{x}$ = position and velocity state (dimension 6)

$\vec{x}_s$ = solve-for parameter state (dimension n_1)

$\vec{u}$ = dynamic consider parameter state (dimension n_2)

$\vec{v}$ = measurement consider parameter state (dimension n_3)

The linearized equations of motion have form

$$\dot{\vec{x}} = F_1\vec{x} + F_2\vec{x}_s + F_3\vec{u}$$

$$\dot{\vec{x}}_s = 0$$

$$\dot{\vec{u}} = 0$$

$$\dot{\vec{v}} = 0$$

and solution

$$\vec{x}_{k+1} = \Phi(k+1,k)\vec{x}_k + \theta_{xx_s}(k+1,k)\vec{x}_{s_k} + \theta_{xu}(k+1,k)\vec{u}_k + \vec{q}_k$$

$$\vec{x}_{s_{k+1}} = \vec{x}_{s_k}$$

$$\vec{u}_{k+1} = \vec{u}_k$$

$$\vec{v}_{k+1} = \vec{v}_k$$

where dynamic noise $\vec{q}_k$ has been added to the solution of $\vec{x}_{k+1}$. This solution can be written in augmented form

$$\vec{x}_{k+1}^A = \Phi^A(k+1,k)\,\vec{x}^A + \vec{q}_k^A$$

where the augmented state transition matrix $\Phi^A(k+1,k)$ is defined as

$$\Phi^A(k+1,k) = \begin{bmatrix} \Phi & \theta_{xx_s} & \theta_{xu} & 0 \\ 0 & I_{n_1 \times n_1} & 0 & 0 \\ 0 & 0 & I_{n_2 \times n_2} & 0 \\ 0 & 0 & 0 & I_{n_3 \times n_3} \end{bmatrix}$$

Henceforth state transition matrix partitions will be written without stating the associated interval of time, which will always be assumed to be $[k, k$.

The measurement deviation vector $\vec{y}$ (dimension m) is related to the augmented deviation state vector through the equation

$$\vec{y}_k = H_k^A \vec{x}_k^A + \vec{\eta}_k$$

where the augmented observation matrix is defined as

$$H_k^A = \begin{bmatrix} H_k & M_k & G_k & L_k \end{bmatrix}$$

and $\vec{\eta}_k$ is measurement noise.

The augmented state covariance matrix P_k^A can be written in terms of its partitions as

$$P_k^A = \begin{bmatrix} P_k & C_{xx_{s_k}} & C_{xu_k} & C_{xv_k} \\ C^T_{xx_{s_k}} & P_{s_k} & C_{x_s u_k} & C_{x_s v_k} \\ C^T_{xu_k} & C^T_{x_s u_k} & U_o & C_{uv_k} \\ C^T_{xv_k} & C^T_{x_s v_k} & C^T_{uv_k} & V_o \end{bmatrix}$$

Prediction and filtering equations for the partitions appearing in the previous equation will be written below. Equations need not be written for the consider parameter covariances U_o and V_o since these do not change with time. Also, C_{uv} will be set to zero because of the assumption that no cross-correlation exists between dynamic and measurement consider parameters. In the equations below Q and R represent the covariances of the dynamic and measurement noises, respectively, defined previously. A minus superscript on covariance partitions indicates the covariance partition immediately prior to processing a measurement; a plus superscript, immediately after processing a measurement. If ICØDE indicates that a measurement is not to be processed, the filtering equations are bypassed. To improve numerical accuracy and avoid non-positive definite covariance matrices, P^-, P_s^-, P^+, P_s^+, J, and J^{-1} are always symmetrized after their computation.

Prediction equations:

$$P_{k+1}^- = (\Phi P_k^+ + \theta_{xx_s} C_{xx_{s_k}}^{+T} + \theta_{xu} C_{xu_k}^{+T}) \Phi^T + C_{xx_{s_{k+1}}}^- \theta_{xx_s}^T + C_{xu_{k+1}}^- \theta_{xu}^T + Q_k$$

$$C_{xx_{s_{k+1}}}^- = \Phi C_{xx_{s_k}}^+ + \theta_{xx_s} P_{s_k}^+ + \theta_{xu} C_{x_s u_k}^{+T}$$

$$P_{s_{k+1}}^- = P_{s_k}^+$$

$$C_{xu_{k+1}}^- = \Phi C_{xu_k}^+ + \theta_{xx_s} C_{x_s u_k}^+ + \theta_{xu} U_o$$

$$C_{x_s u_{k+1}}^- = C_{x_s u_k}^+$$

$$C_{xv_{k+1}}^- = \Phi C_{xv_k}^+ + \theta_{xx_s} C_{x_s v_k}^+$$

$$C_{x_s v_{k+1}}^- = C_{x_s v_k}^+$$

Filtering equations:

$$J_{k+1} = H_{k+1} A_{k+1} + M_{k+1} B_{k+1} + G_{k+1} D_{k+1} + L_{k+1} E_{k+1} + R_{k+1}$$

where

$$A_{k+1} = P^-_{k+1} H^T_{k+1} + C^-_{xx_{s_{k+1}}} M^T_{k+1} + C^-_{xu_{k+1}} G^T_{k+1} + C^-_{xv_{k+1}} L^T_{k+1}$$

$$B_{k+1} = P^-_{s_{k+1}} M^T_{k+1} + C^{-T}_{xx_{s_{k+1}}} H^T_{k+1} + C^-_{x_s u_{k+1}} G^T_{k+1} + C^-_{x_s v_{k+1}} L^T_{k+1}$$

$$D_{k+1} = C^{-T}_{xu_{k+1}} H^T_{k+1} + C^{-T}_{x_s u_{k+1}} M^T_{k+1} + U_o G^T_{k+1}$$

$$E_{k+1} = C^{-T}_{xv_{k+1}} H^T_{k+1} + C^{-T}_{x_s v_{k+1}} M^T_{k+1} + V_o L^T_{k+1}$$

The Kalman gain matrices for both position/velocity state and solve-for parameters are given by

$$K_{k+1} = A_{k+1} J^{-1}_{k+1}$$

$$S_{k+1} = B_{k+1} J^{-1}_{k+1}$$

The covariance partitions immediately after processing a measurement are given by

$$P^+_{k+1} = P^-_{k+1} - K_{k+1} A^T_{k+1}$$

$$C^+_{xx_{s_{k+1}}} = C^-_{xx_{s_{k+1}}} - K_{k+1} B^T_{k+1}$$

$$P^+_{s_{k+1}} = P^-_{s_{k+1}} - S_{k+1} B^T_{k+1}$$

$$C^+_{xu_{k+1}} = C^-_{xu_{k+1}} - K_{k+1} D^T_{k+1}$$

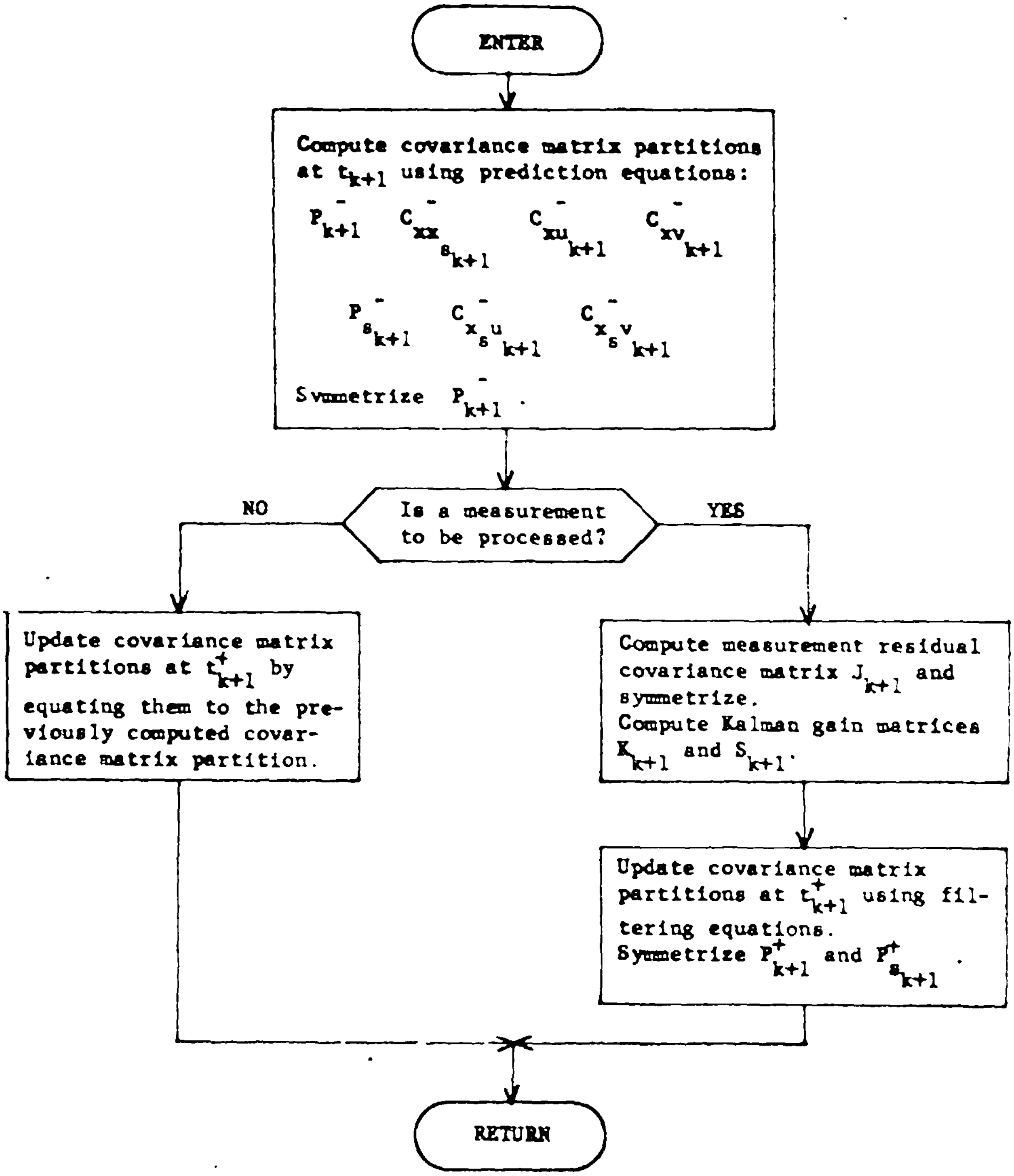
$$C^{+}_{x_s u_{k+1}} = C^{-}_{x_s u_{k+1}} - S_{k+1} D^{T}_{k+1}$$

$$C^{+}_{xv_{k+1}} = C^{-}_{xv_{k+1}} - K_{k+1} E^{T}_{k+1}$$

$$C^{+}_{x_s v_{k+1}} = C^{-}_{x_s v_{k+1}} - S_{k+1} E^{T}_{k+1}$$

Reference: Lee, Gentry; Falce, Ralph; and Hopper, Fred: Interplanetary Trajectory Error Analysis, Volume I - Analytical Manual. MCR-67-441, Martin Marietta Corporation, Denver, Colorado, December 1967 (Completed under Contract NAS8-21120).

NAVM Flow Chart



SUBROUTINE NDTM

PURPOSE: TO COMPUTE THE UNAUGMENTED PORTION OF THE STATE TRANSITION MATRIX USING THE NUMERICAL DIFFERENCE TECHNIQUE.

CALLING SEQUENCE: CALL NDTM(RI,RF)

ARGUMENT:	RF	I	POSITION AND VELOCITY OF THE VEHICLE AT THE END OF THE TIME INTERVAL
	RI	I	POSITION AND VELOCITY OF THE VEHICLE AT THE BEGINNING OF THE TIME INTERVAL

SUBROUTINES SUPPORTED: PSIM

SUBROUTINES REQUIRED: NTM

LOCAL SYMBOLS:	F1	TEMPORARY STORAGE FOR FACP
	F2	TEMPORARY STORAGE FOR FACV
	IPR	INTERMEDIATE STORAGE FOR IPRINT
	RP	POSITION AND VELOCITY OF VEHICLE AT INITIAL TIME
	SAVE	TEMPORARY STORAGE FOR ACC
	T	ALTERED POSITION AND VELOCITY OF VEHICLE AT INITIAL TIME
	U	ALTERED POSITION AND VELOCITY OF VEHICLE AT FINAL TIME

COMMON COMPUTED/USED: ACC FACP FACV IPRINT

COMMON COMPUTED: PHI

COMMON USED: ACCND DELTM NDACC ONE ZERO

NDTM Analysis

The nonlinear equations of motion of the spacecraft can be written symbolically as

$$\dot{\vec{x}} = \vec{f}\,(\vec{x},\, t\,) \tag{1}$$

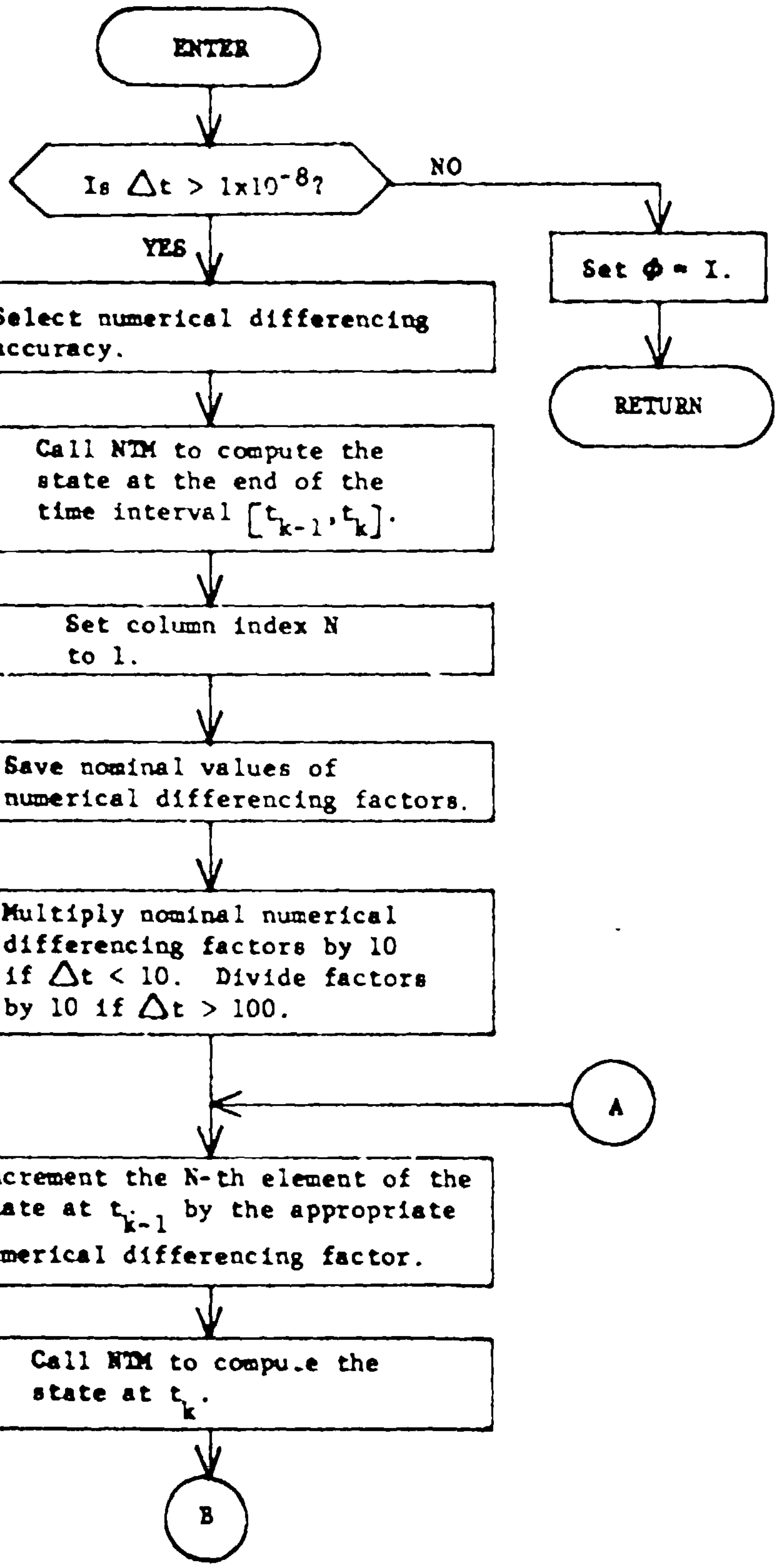
where $\vec{x}$ is the spacecraft position/velocity state.

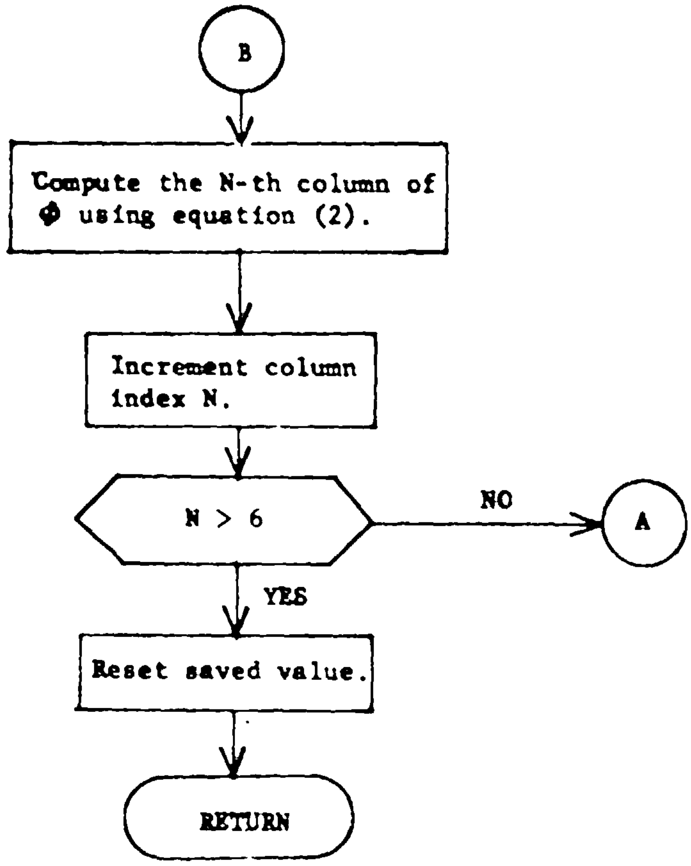
Suppose we wish to use numerical differencing to compute the state transition matrix $\Phi(t_k, t_{k-1})$. Let $\vec{\Phi}(t_k, t_{k-1})$ represent the j-th column of $\Phi_j(t_k, t_{k-1})$. We assume we have available the nominal states $\vec{x}^*(t_{k-1})$ and $\vec{x}^*(t_k)$. To obtain $\vec{\Phi}_j(t_k, t_{k-1})$ we increment the j-th element of $\vec{x}^*(t_{k-1})$ by the numerical differencing factor Δx_j and numerically integrate equation (1) over the time interval $\left[t_{k-1},\, t_k\right]$ to obtain the new spacecraft state $\vec{x}_j(t_k)$. The j-subscript indicates $\vec{x}_j(t_k)$ was obtained by incrementing the j-th element of $\vec{x}^*(t_{k-1})$. Then

$$\vec{\Phi}_j(t_k,\, t_{k-1}) = \frac{\vec{x}_j(t_k) - \vec{x}^*(t_k)}{\Delta x_j} \tag{2}$$

$$j = 1,2,\ldots,\, 6$$

NDTM Flow Chart




B
Compute the N-th column of Φ using equation (2).
Increment column index N.
N > 6
NO
A
YES
Reset saved value.
RETURN

SUBROUTINE NEWPGE

PURPOSE: PRINTS APPROPRIATE HEADING AT THE TOP OF EACH PAGE WHEN PRINTOUT OF TRAJECTORY INFORMATION IS DESIRED

CALLING SEQUENCE CALL NEWPGE

SUBROUTINES SUPPORTED INPUTZ PRINT SPACE

SUBROUTINES REQUIRED: NONE

COMMON COMPUTED/USED: IPG

COMMON COMPUTED: LINCT

COMMON USED: KL

PROGRAM NOMNAL

PURPOSE: TO CONTROL THE ENTIRE GENERATION OF A NOMINAL TRAJECTORY FROM INJECTION TARGETING THROUGH MIDCOURSE CORRECTIONS AND ORBIT INSERTION.

CALLING SEQUENCE: NONE (MAIN PROGRAM)

SUBROUTINES SUPPORTED: NONE (MAIN PROGRAM)

SUBROUTINES REQUIRED: GIDANS PRELIM TRJTRY

COMMON COMPUTED: IPRE

COMMON USED: KWIT

NOMNAL Analysis

NOMNAL is the executive program controlling the entire generation of a nominal trajectory from injection targeting through midcourse corrections and orbit insertion.

NOMNAL begins by calling PRELIM for the preliminary work including initialization of variables, reading of the input data, and computation of zero iterate values of initial time, position, and velocity if required.

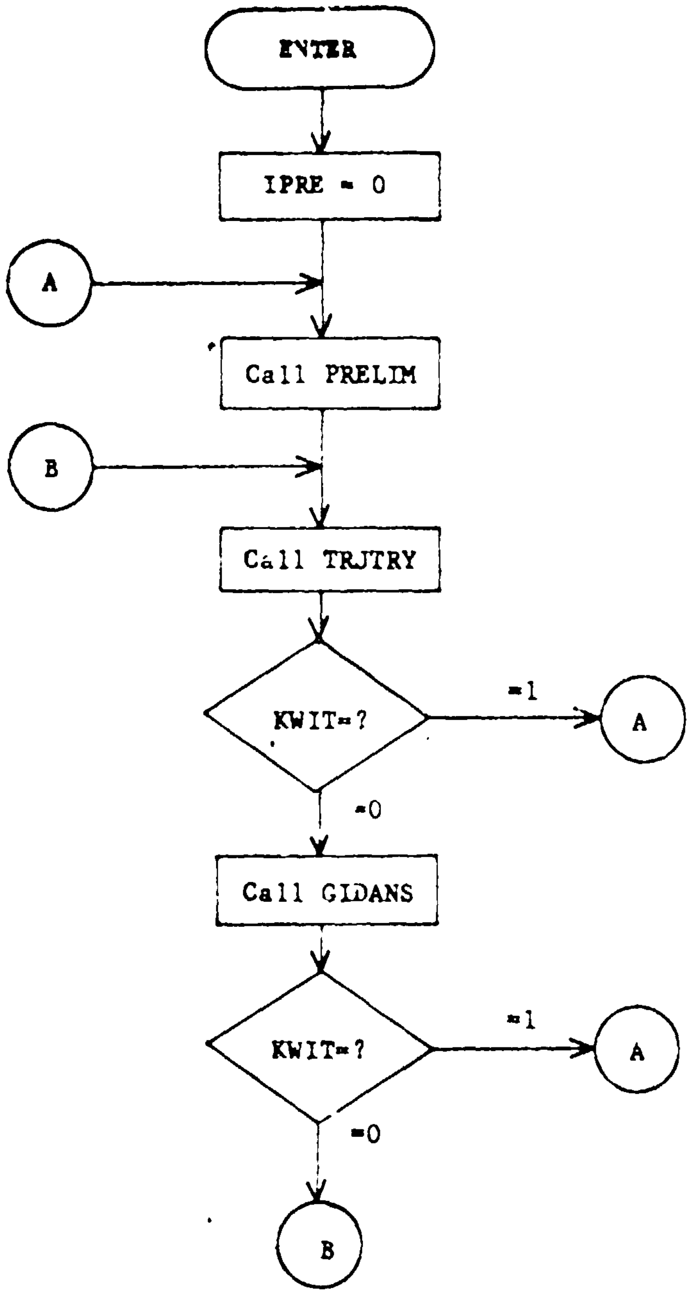
NOMNAL then calls TRJTRY. TRJTRY first determines the time of the next guidance event. It then integrates and records the nominal trajectory to that time. TRJTRY then returns control to NOMNAL.

NOMNAL next calls GIDANS. GIDANS processes the computation and execution of the current guidance event. NOMNAL then reenters its basic cycle by calling TRJTRY to propagate the corrected trajectory to its next guidance event.

Two flags are used by NOMNAL. The flag IPRE is initialized at zero in NOMNAL. During the processing of the first data case PRELIM sets it to unity. PRELIM uses IPRE to determine whether to preset constants to internally stored values or leave them at their previous values before reading the next data case.

The second flag KWIT determines whether the current case should be continued or terminated according to the flag value zero or unity respectively. Termination is indicated when a fatal error occurs during trajectory propagation or guidance event computation or when the desired end time is reached.

NOMNAL Flow Chart



SUBROUTINE NONINS

PURPOSE: TO DETERMINE THE TIME AND CORRECTION VECTOR FOR AN INSERTION FROM AN APPROACH HYPERBOLA INTO A SPECIFIED PLANE AND AS NEAR AS POSSIBLE TO A PRESCRIBED CLOSED ORBIT.

CALLING SEQUENCE: CALL NONINS(GM,X,Z,DA,DE,DWTP,DI,DN,TMEX,VEL, IEX)

ARGUMENT:	GM	I	GRAVITATIONAL CONSTANT
	X(3)	I	POSITION VECTOR AT DECISION
	Z(3)	I	VELOCITY VECTOR AT DECISION
	DA	I	DESIRED SEMIMAJOR AXIS
	DE	I	DESIRED ECCENTRICITY
	DWTP	I	DESIRED ARGUMENT OF PRRIAPSIS
	DWTP	I	DESIRED ARGUMENT OF PERIAPSIS
	DI	I	DESIRED INCLINATION
	DN	I	DESIRED LONGITUDE OF ASCENDING NODE
	TMEX	O	TIME FROM DECISION TO EXECUTION (SECONDS)
	VEL(3)	O	INSERTION VELOCITY VECTOR
	IEX	O	EXECUTION CODE =0 EXECUTABLE SOLUTION DETERMINED =1 NO EXECUTABLE SOLUTION FOUND

SUBROUTINES SUPPORTED: INSERS

SUBROUTINES REQUIRED: CARELL ELCAR

LOCAL SYMBOLS:	AH	HYPERBOLIC SEMIMAJOR AXIS
	ANG	TRUE ANOMALY OF HYPERBOLIC ASYMPTOTE
	ARC2	360.
	ARC	180.
	A	SEMIMAJOR AXIS OF MODIFIED ELLIPSE
	CEI	COSINE OF DI
	CEN	COSINE OF DN

CHI	COSINE OF HI
CHN	COSINE OF HN
CTAE	COSINE OF ETA
CTASY	COSINE OF TASY
CWTXE	COSINE OF WTXE
CWTXH	COSINE OF WTXH
DELV	VELOCITY CORRECTIONS OF CANDIDATE SOLUTION
DRA	DESIRED APOAPSIS RADIUS
DRP	DESIRED PERIAPSIS RADIUS
DTA	DUMMY VARIABLE FOR OUTPUT
DVM	MAGNITUDES OF CANDIDATE CORRECTIONS
DV	MAGNITUDES OF CANDIDATE CORRECTIONS
EH	HYPERBOLIC ECCENTRICITY
ERRMAX	SCALAR ERROR ASSIGNED TO IMPOSSIBLE SOLUTION
ERR	ARRAY OF SCALAR ERRORS OF SOLUTIONS
ETAX	TRUE ANOMALIES AT INTERSECTION POINTS ON ELLIPSE
E	ECCENTRICITY OF MODIFIED ELLIPSE
HI	HYPERBOLIC INCLINATION
HN	HYPERBOLIC LONGITUDE OF ASCENDING NODE
HRP	HYPERBOLIC PERIAPSIS RADIUS
HTAX	TRUE ANOMALIES AT INTERSECTION POINTS ON HYPERBOLA
HTA	CANDIDATE HYPERBOLIC TRUE ANOMALY AT INTERSECTION
MININ	INDEX OF OPTIMAL INTERSECTION POINT
MIN	INDEX OF OPTIMAL SOLUTION (POINT AND MOD)

NCPOS	FLAG INDICATING WHETHER ANGLE BETWEEN NODE AND INTERSECTION IS GREATER OR LESS THAN 180
NSOLS	NUMBER OF SOLUTIONS
NT1	INDEX OF FIRST SOLUTION
NT2	INDEX OF LAST SOLUTION
PH	HYPERBOLIC SEMILATUS RECTUM
PI	THE MATHEMATICAL CONSTANT PI
PP	UNIT VECTOR TOWARD PERIAPSIS
QQ	UNIT VECTOR IN ORBITAL PLANE NORMAL TO PP
RAD	DEGREE TO RADIAN FACTOR
RA	APOAPSIS RADIUS
RHYP	HYPERBOLIC CANDIDATE RADII TO INTERSECTION
RMAG	RADIUS TO INTERSECTION POINT
RM	RADIUS AT DECISION
RP	PERIAPSIS RADIUS
RX	RADIUS TO INTERSECTION POINT ON ELLIPSE
R1	RADIUS VECTOR TO HYPERBOLA AT INTERSECTION
R	RADIUS VECTOR TO ELLIPSE AT INTERSECTION
SEI	SINE OF DI
SEN	SINE OF DN
SGNZ	SIGN OF DECLINATION OF INTERSECTION POINT
SHI	SINE OF HI
SHN	SINE OF HN
STA	TRUE ANOMALY AT DECISION
TAE	ARRAY OF ELLIPTIC TRUE ANOMALIES
TASY	TRUE ANOMALY OF ASYMPTOTE
TAXE	TRUE ANOMALY AT INTERSECTION POINT ON

	ELLIPSE
TAXH	TRUE ANOMALY AT INTERSECTION POINT ON HYPERBOLA
TA	TRUE ANOMALY
TDEC	TIME FROM PERIAPSIS AT DECISION
TEXC	TIME FROM PERIAPSIS AT EXECUTION ON HYPER-BOLA
TEX	ARRAY OF CANDIDATE TIMES FROM DECISION TO EXECUTION
TTF	TIME FROM PERIAPSIS AT DECISION ON ELLIPSE
VM	SPEED AT DECISION
V1	VELOCITY VECTOR ON HYPERBOLA AT EXECUTION
V	VELOCITY VECTOR ON ELLIPSE AT EXECUTION
WE	ARGUMENT OF PERIAPSIS ON ELLIPSE
WH	ARGUMENT OF PERIAPSIS ON HYPERBOLA
WTXE	ANGLE BETWEEN ASCENDING NODE AND INTER-SECTION POINT ON ELLIPSE
WTXH	ANGLE BETWEEN ASCENDING NODE AND INTER-SECTION POINT ON HYPERBOLA
WW	UNIT NORMAL TO ORBITAL PLANE
W	ARGUMENT OF PERIAPSIS
XINT	X COMPONENT OF INTERSECTION POINT
YINT	Y COMPONENT OF INTERSECTION POINT
ZINT	Z COMPONENT OF INTERSECTION POINT

NONINS Analysis

NONINS determines the time and correction vector for an impulsive insertion from an approach hyperbola into a specified plane and as near as possible to a prescribed closed orbit. The approach hyperbola is specified by giving the planetocentric equatorial state $\vec{r}, \vec{v}$ at the time of decision t_d. The final orbit is defined by giving its desired orbital elements $(a_E, e_E, i_E, \omega_E, \Omega_E)$ again in planetocentric equatorial coordinates.

Subroutine CAREL is first called to convert the hyperbolic state at decision $\vec{r}, \vec{v}$ into Keplerian conic elements $(a_H, e_H, i_H, \omega_H, \Omega_H, t_{Hd})$ where t_{Hd} is the time from periapsis at decision (negative on the approach ray).

The points of intersection of the approach orbital plane and the desired orbital plane are then determined. The elements defining the two planes are therefore given by i_H, Ω_H and i_E, Ω_E. Let $\hat{A}$ denote the unit vector toward the ascending node of an orbit and $\hat{B}$ denote the in-plane normal to $\hat{A}$ in the direction of motion. Then

$$\hat{A} = (\cos\Omega\ ,\ \sin\Omega\ ,\ 0) \tag{1}$$

$$\hat{B} = (-\sin\Omega \cos i,\ \cos\Omega \cos i,\ \sin i) \tag{2}$$

Hence the normal to the orbital plane $\hat{C}$ is given by $\hat{C} = \hat{A} \times \hat{B}$ or

$$\hat{C} = (\sin\Omega \sin i,\ -\cos\Omega \sin i,\ \cos i) \tag{3}$$

The direction of the line of intersection of the two planes is therefore determined by $\vec{X} = \hat{C}_H \times \hat{C}_E$ or

$$\begin{aligned}\vec{X} = (&\cos i_H \sin i_E \cos\Omega_E - \sin i_H \cos i_E \cos\Omega_H,\\ &\cos i_H \sin i_E \sin\Omega_E - \sin i_H \cos i_E \sin\Omega_H,\\ &\sin i_H \sin i_E (\cos\Omega_H \sin\Omega_E - \sin\Omega_H \cos\Omega_E))\end{aligned} \tag{4}$$

Then the unit vector along the line of intersection toward the northern hemisphere is given by

$$\hat{X} = \operatorname{sgn} \vec{X}_3 \frac{\vec{X}}{X} \tag{5}$$

Therefore the true anomaly f_{HX} along the hyperbola at the northern intersection point is given by

$$\cos(\omega_H + f_{HX}) = \hat{X} \cdot \hat{A}_H \tag{6}$$

The true anomaly on the hyperbola at the southern point is therefore $f_{HX} + 180^\circ$. Note that there exists a region of true anomalies lying between the incoming and outgoing asymptotes for which the hyperbola is not defined. Similar equations define the true anomaly on the ellipse at the two points of intersection. Note that this implies that the modified ellipse will have the same ω as the desired ellipse.

For the intersection true anomaly f_{HX} the radius magnitude on the hyperbola may be determined

$$r_I = \frac{a_H(1 - e_H^2)}{1 + e_H \cos f_H} \tag{7}$$

To permit an impulsive insertion, a_E and e_E must be modified to satisfy

$$r_I = \frac{a_E(1 - e_E^2)}{1 + e_E \cos f_E} \tag{8}$$

There are three candidate modifications examined to determine a "best" one: (1) Vary r_a while holding r_p constant

(2) Vary r_p while holding r_a constant

(3) Vary a while holding e constant

"Best" is defined below in terms of a weighted scalar function of the changes in r_a and r_p.

Rewriting (8) in terms of r_a and r_p (using $a = \frac{r_a + r_p}{2}$, $e = \frac{r_a - r_p}{r_a + r_p}$) yields the useful relation

$$r_a(1 + \cos f_E) + r_p(1 - \cos f_E) = \frac{2 r_a r_p}{r_I} \tag{9}$$

Equation (9) may be solved for r_a as

$$r_a = \frac{r_I r_p (1 - \cos f_E)}{2 r_p - r_I (1 + \cos f_E)} \tag{10}$$

This yields the r_a which defines the modified orbit holding r_p at its desired value. The semi-major axis and eccentricity are then computed from $a = \frac{r_a + r_p}{2}$, $e = \frac{r_a - r_p}{r_a + r_p}$.

Similarly (9) may be solved for r_p as

$$r_p = \frac{r_I r_a (1 + \cos f)}{2\, r_a - r_I(1 - \cos f)} \qquad (11)$$

This determines the modification is r_p required to achieve an intersecting ellipse having the desired r_a.

Finally (8) may be solved trivially for the a_E required to produ intersection for the desired eccentricity.

$$a_E = \frac{r_I(1 + e_E \cos f_E)}{(1 - e_E^2)} \qquad (12)$$

An error is assigned to each of the candidate solutions as

$$E_i = W_i \left[|\Delta r_a| + |\Delta r_p| \right]$$

where Δr_a, Δr_p are the errors between the desired and modified values of r_a and r_p. The weighting factor W_i is assigned rather arbitrarily. Currently the weighting factor is $W_i = w_{1i}\, w_{2i}$ where

$w_{1i} =$ 1 if the true anomaly is on the incoming ray
2 if the true anomaly is on the outgoing ray

$w_{2i} =$ 1 if option 1
2 if option 2
3 if option 3

Thus a solution on the incoming asymptote is preferred over one on the outgoing asymptote and one subsequent trim is preferred over two subsequent trims.

Having determined the elements of an intersecting orbit the insertion parameters are easily computed. The velocity on the hyperbola at the intersection point may be computed from ELCAR as $\vec{v}_H$. The velocity on the ellipse following the insertion is computed by calling ELCAR with the modified elliptical element to get $\vec{v}_E$. The impulsive $\overrightarrow{\Delta v}$ is then given by

$$\overrightarrow{\Delta v} = \vec{v}_E - \vec{v}_H$$

The time interval from th ecision to the execution is given by the hyperbolic time from the initial point to the relevant intersection point.

NONINS Flow Chart

ENTER

Compute hyperbolic elements at time of decision (CAREL) and record.

Compute true anomaly on hyperbolic at two points of intersection with desired orbital plane f_{h1}, f_{h2}.

Eliminate any solution occurring before time of decision or in impossible region between two asymptotes and set indices NT1, NT2, NSOLS accordingly.

I = NT1

A

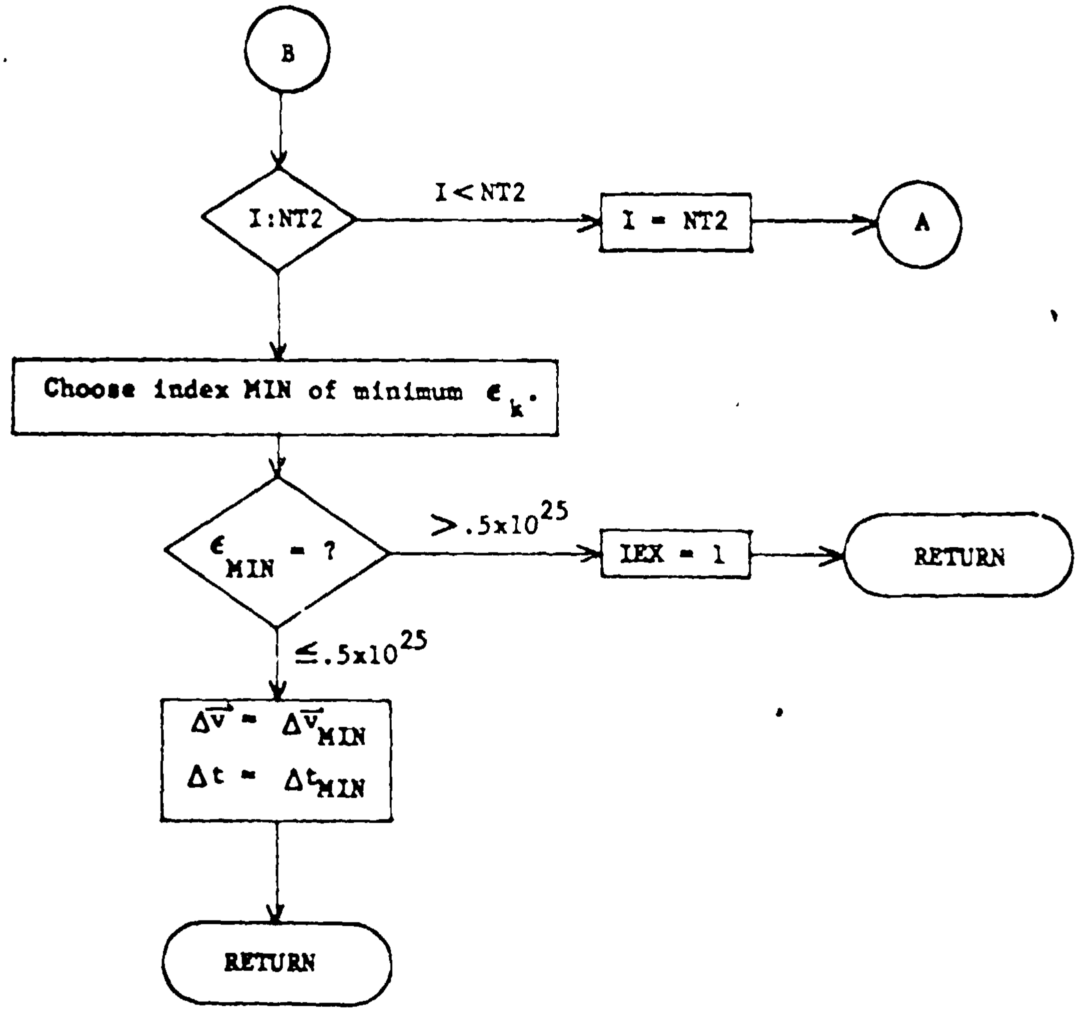
Compute cartesian state on hyperbola at f_{hi} (ELCAR) and record.

Modify r_a of desired ellipse to obtain intersection with hyperbola and compute cartesian state (ELCAR). Compute $\Delta\vec{v}_1$ and $\epsilon_1 = 2I \cdot |\Delta r_a|$.

Modify r_p of desired ellipse to obtain intersection with hyperbola and compute cartesian state (ELCAR). Compute $\Delta\vec{v}_2$ and $\epsilon_2 = I \cdot |\Delta r_p|$.

Modify "a" of desired ellipse to obtain intersection with hyperbola and compute cartesian state (ELCAR). Compute $\Delta\vec{v}_3$ and $\epsilon_3 = 3I\ (|\Delta r_a + \Delta r_p|)$.

B


B
I:NT2
I < NT2
I = NT2
A
Choose index MIN of minimum ϵ_k.
ϵ_{MIN} = ?
$> .5 \times 10^{25}$
IEX = 1
RETURN
$\leq .5 \times 10^{25}$
$\Delta\vec{v} = \Delta\vec{v}_{MIN}$
$\Delta t = \Delta t_{MIN}$
RETURN

SUBROUTINE NONLIN

PURPOSE: TO CONTROL EXECUTION OF NON-LINEAR GUIDANCE EVENTS.

CALLING SEQUENCE: CALL NONLIN

SUBROUTINES SUPPORTED: GUISIM GUIDM

SUBROUTINES REQUIRED: CAREL ELCAR GIDANS

LOCAL SYMBOLS:

AA	ARGUMENT FOR SUBROUTINE CAREL
DI	JULIAN DATE OF EVENT
EE	ARGUMENT FOR SUBROUTINE CAREL
ISNPR	SAVE INPR VALUE
ISPRNT	SAVE IPRINT VALUE
KEY	INTERMEDIATE VARIABLE IN SETTING UP TARGET ARRAY
KICL2	SAVE ICL2 VALUE
KICL	SAVE ICL VALUE
KISPH	SAVE ISPH VALUE
KISP2	SAVE ISP2 VALUE
ODELT	SAVE ORIGINAL DELTP VALUE
OSPH	SAVE ORIGINAL SPHERE OF INFLUENCE OF TARGET PLANET
PP	ARGUMENT RETURNED FROM CAREL
QQ	ARGUMENT RETURNED FROM CAREL
RMAG	ARGUMENT FOR SUBROUTINE ELCAR
TAA	TRUE ANOMALY
TFFP	TIME OF FLIGHT FROM PERIAPSIS
TFP	TIME FROM PERIAPSIS
TRTIME	TRAJECTORY TIME OF THE GUIDANCE EVENT
VMAG	ARGUMENT FOR SUBROUTINE ELCAR
WW	ARGUMENT FOR SUBROUTINE CAREL

XXI	ARGUMENT FOR SUBROUTINE CAREL
XXN	ARGUMENT FOR SUBROUTINE CAREL
XYZTAA	ZERO TRUE ANOMALY ARGUMENT FOR SUBROUTINE ELCAR

COMMON COMPUTED/USED:	DELTP	DELV	DSI	DT	IBADS
	ICL2	ICL	INPR	IPRINT	ISPH
	ISP2	KLP	KMXQ	KTAR	KTIM
	KTP	KWIT	MAT	MAXB	MOL
	NOIT	NTP	RSI	SPHERE	TAR
	TGT3	TIN	TMU	TOL	VSI
	XDC	XDELV	XRC	ZDAT	
COMMON COMPUTED:	ACKT	AC	BDR	BDT	DC
	DG	DVMAX	D1	ISTART	IZERO
	KTYP	KUR	LVLS	NLP	NOGYD
	NPAR	PERV	RC	RIN	SPHFAC
	TIMG	TRTM			
COMMON USED:	ACX	ALNGTH	DATEJ	DELTAV	IX
	JX	LKLP	LKTAR	LKTP	LLVLS
	LNPAR	PMASS	TM	T3	XAC
	XBDR	XBDT	XDSI	XDVMAX	XFAC
	XIN	XPERV	XRSI	XTAR	XTOL
	XVSI	ZERO			

NØNLIN Analysis

NØNLIN is the interface subroutine between the non-linear guidance subroutines of NØMNAL and subroutine GUIDM of ERRAN and subroutine GUISIM of SIMUL. NØNLIN selects the necessary data from the ERRAN and SIMUL common blocks and stores into the common blocks of NØMNAL the information needed to compute and/or execute the $\Delta\vec{V}$ required in order to meet specified target conditions.

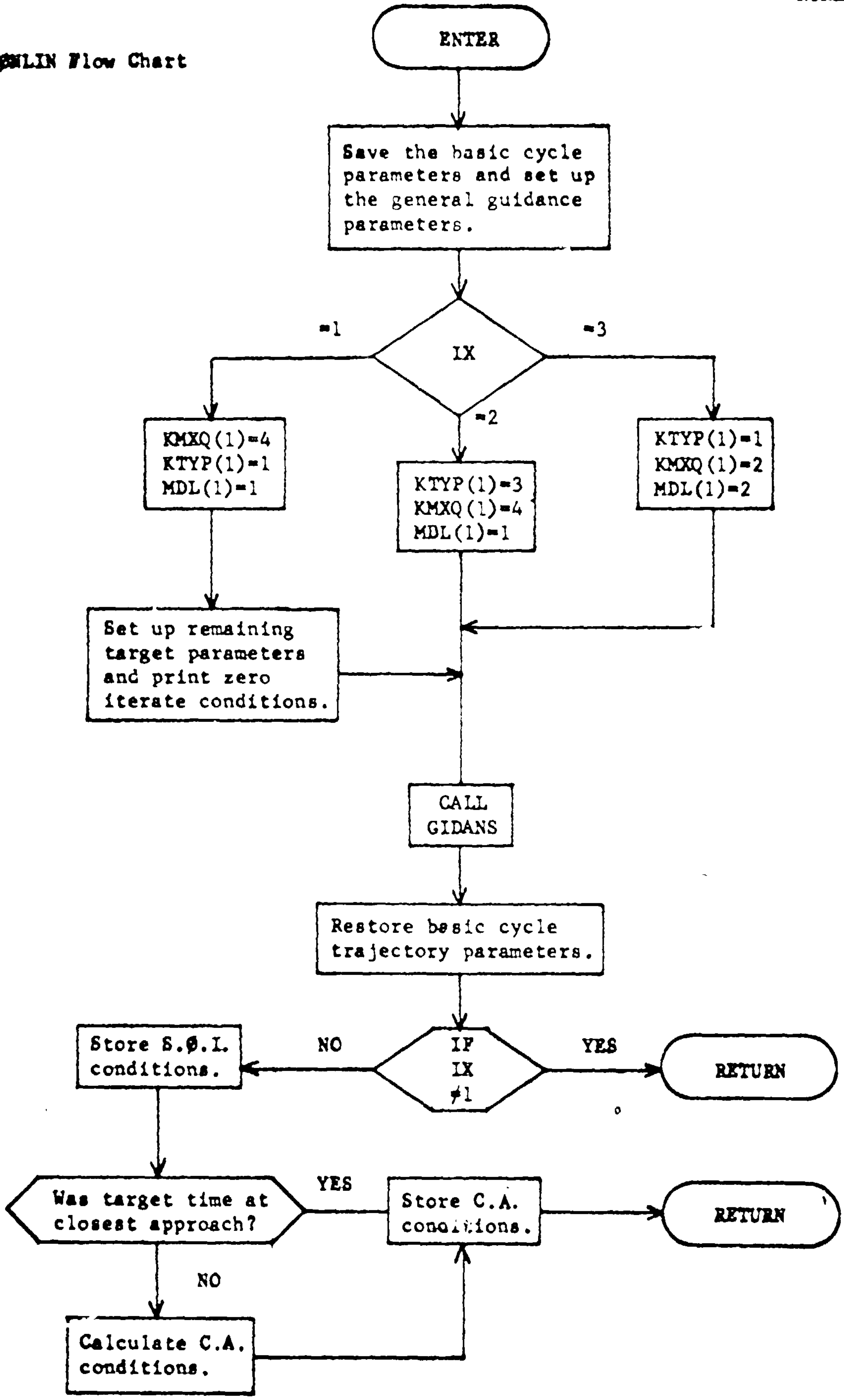
The most important task performed by NØNLIN is the selection of the desired guidance scheme. The variable IX is tested and control is transformed according to the following:

IX = 1, retargeting to specified target parameters
   = 2, bit insertion to specified orbit
   = 3, $\Delta\vec{V}$ execution by a series of specified pulses

For each type of event, NØNLIN then sets up values controlling the type of guidance event (KTYP), implementation code (KMXQ), and execution model code (MDL). For retargeting only, NØNLIN stores the remaining values needed for $\Delta\vec{V}$ calculation and prints the zero iterate conditions.

NØMNAL calls GIDANS to perform the guidance event and restores parameters necessary for the basic cycles of ERRAN and SIMUL. For retargeting only, NØMNAL then stores the conditions at sphere of influence and closest approach of the target planet which were calculated by subroutine TARGET.

NØNLIN Flow Chart



SUBROUTINE NTM

PURPOSE: CONTROL COMPUTATION OF NOMINAL TRAJECTORY IN THE ERROR ANALYSIS PROGRAM

CALLING SEQUENCE: CALL NTM(RI,RF,NTMC,ICODE)

ARGUMENT:	ICODE	I	INTERNAL CODE THAT DETERMINES WHICH TRAJECTORY IS BEING RUN AND WHAT INFORMATION IS DESIRED
	NTMC	I	NOMINAL TRAJECTORY MODULE CODE THAT DETERMINES WHICH TYPE OF TRAJECTORY PROGRAM IS TO BE USED (NOTE ONLY THE VIRTUAL MASS TECHNIQUE IS SUPPLIED WITH THIS PROGRAM. HOWEVER, WITH LITTLE EFFORT ANY TRAJECTORY PROGRAM MAY BE ADDED AS AN EXTRA OPTION.)
	RF	O	POSITION AND VELOCITY OF THE VEHICLE AT THE END OF THE TIME INTERVAL
	RI	I	POSITION AND VELOCITY OF THE VEHICLE AT THE BEGINNING OF THE TIME INTERVAL

SUBROUTINES SUPPORTED: ERRANN MUND NDTM PLND PSIM SETEVN GUID VARADA PRED

SUBROUTINES REQUIRED: VMP

LOCAL SYMBOLS:	D1	JULIAN DATE, EPOCH JAN.0, 1900, OF INITIAL TRAJECTORY TIME
	RMP	DISTANCE OF VEHICLE FROM TARGET PLANET AT SPHERE OF INFLUENCE OR CLOSEST APPROACH
	VMM	MAGNITUDE OF THE VELOCITY VECTOR

COMMON COMPUTED/USED:	BDRSI1	BDRSI2	BDRSI3	BDTSI1	BDTSI2
	BDTSI3	BSI1	BSI2	BSI3	ICA1
	ICA2	ICA3	ICL	ISOI1	ISOI2
	ISOI3	ISPH	RCA1	RCA2	RCA3
	RSOI1	RSOI2	RSOI3	TCA1	TCA2
	TCA3	TSOI1	TSOI2	TSOI3	VSOI1
	VSOI2	VSOI3			

COMMON USED:	ACC	BDR	BDT	B	DATEJ
	DC	DELTM	DSI	IPROB	ISP2
	ITR	NQE	RC	RSI	TRTM1

NTM Analysis

Subroutine NTM is used to generate the (most recent) targeted nominal trajectory in the error analysis mode. Subroutine NTM is equivalent to a subroutine NTMS from which all loops associated with ICODE = -3, -2, 2, 3 have been removed. For this reason no further analysis and no flow chart will be presented for subroutine NTM. Refer to subroutine NTMS.

SUBROUTINE NTMS

PURPOSE: CONTROL COMPUTATION OF TARGETED NOMINAL, MOST RECENT NOMINAL, AND ACTUAL TRAJECTORIES IN THE SIMULATION PROGRAM

CALLING SEQUENCE: CALL NTMS(RI,RF,NTMC,ICODE)

ARGUMENT:

ICODE	I	INTERNAL CODE THAT DETERMINES WHICH TRAJECTORY IS BEING RUN AND WHAT INFORMATION IS DESIRED
NTMC	I	NOMINAL TRAJECTORY MODULE CODE THAT DETERMINES WHICH TYPE OF TRAJECTORY PROGRAM IS TO BE USED (NOTE ONLY THE VIRTUAL MASS TECHNIQUE IS SUPPLIED WITH THIS PROGRAM. HOWEVER, WITH LITTLE EFFORT ANY TRAJECTORY PROGRAM MAY BE ADDED AS AN EXTRA OPTION.)
RF	O	POSITION AND VELOCITY OF THE VEHICLE AT THE END OF THE TIME INTERVAL
RI	I	POSITION AND VELOCITY OF THE VEHICLE AT THE BEGINNING OF THE TIME INTERVAL

SUBROUTINES SUPPORTED: SIMULL MUND NDTM PLND PSIM SETEVS GUISS VARSIM PRESIM

SUBROUTINES REQUIRED: VMP

LOCAL SYMBOLS:

ACCS	INTERMEDIATE STORAGE FOR ACCURACY
D1	JULIAN DATE, EPOCH JAN.,1900, OF INITIAL TRAJECTORY TIME
K1	INDEX FOR SEMIMAJOR AXIS ELEMENT
K2	INDEX FOR ECCENTRICITY ELEMENT
K3	INDEX FOR INCLINATION ELEMENT
K4	INDEX FOR ASCENDING NODE ELEMENT
K5	INDEX FOR PERIAPSIS ELEMENT
K6	INDEX FOR MEAN ANOMALY ELEMENT
NBODS	INTERMEDIATE STORAGE FOR NBOD
NBS	INTERMEDIATE STORAGE FOR NB ARRAY
RMP	DISTANCE OF VEHICLE FROM TARGET PLANET

SAVE10	INTERMEDIATE STORAGE
SAVE1	INTERMEDIATE STORAGE
SAVE2	INTERMEDIATE STORAGE
SAVE3	INTERMEDIATE STORAGE
SAVE4	INTERMEDIATE STORAGE
SAVE5	INTERMEDIATE STORAGE
SAVE6	INTERMEDIATE STORAGE
SAVE7	INTERMEDIATE STORAGE
SAVE8	INTERMEDIATE STORAGE
VMN	MAGNITUDE OF THE VELOCITY VECTOR

COMMON COMPUTED/USED:	ACC	BDRSI1	BDRSI2	BDRSI3	BDTSI1
	BDTSI2	BDTSI3	BSI1	BSI2	BSI3
	CN	EMN	ICA1	ICA2	ICA3
	ICL	ISOI1	ISOI2	ISOI3	ISPH
	NBOD	NB	PMASS	RCA1	RCA2
	RCA3	RSOI1	RSOI2	RSOI3	SMJR
	ST	TCA1	TCA2	TCA3	TSOI1
	TSOI2	TSOI3	VSOI1	VSOI2	VSOI3
COMMON USED:	ACC1	ALNGTH	BDR	BDT	B
	DAB	DATEJ	DC	DEB	DELTM
	DIB	DMAB	DMUPB	DMUSB	DNOB
	DSI	DWB	IPROB	ISP2	ITR
	NBOD1	NB1	NGE	NQE	NTP
	RC	RSI	TM	TRTM1	VSI

NTMS Analysis

Subroutine NTMS is used to generate any of the three trajectories required in the simulation mode -- the (most recent) targeted nominal trajectory, the most rec'`t nominal trajectory, and the actual trajectory.

The input variable ICODE is used to distinguish between these trajectories. It is unimportant to the virtual mass technique which trajectory is being computed. However, it is important to keep them separated so that the proper codes are set that check for approaching the sphere of influence of the target planet and reaching closest approach. It is also important to keep separate the conditions at which these occur for each trajectory. The following list describes ICODE completely.

ICODE = 3, NTMS will check to see if the sphere of influence and/or closest approach has been reached on the actual trajectory. If not, VMP will check for these conditions and on encountering either, NTMS places the conditions in special storage locations so they will be saved for future reference.

ICODE = 2, NTMS performs the same operations as described above for the most recent nominal trajectory.

ICODE = 1, NTMS again checks for sphere of influence and closest approach as above for the targeted nominal trajectory.

ICODE = 0, the only important information in this situation is the state vector at the end of the time interval. Therefore, NTMS does not check to see if closest approach or sphere of influence is encountered. This might occur in numerical differencing, for example.

ICODE = -1, it is important to know if sphere of influence or closest approach is reached on the targeted nominal trajectory. However, it is not desired that the information be stored for future use. This situation occurs in the guidance event.

ICODE = -2, the same comments may be made as if ICODE = -1, except this is on the most recent nominal trajectory.

ICODE = -3, again, this value of ICODE is treated the same as is ICODE = -1, for the actual trajectory.

Physical constants, planetary ephemerides, and other information relating to the dynamic model are the same for the targeted and most recent nominal trajectories. This is not true for the actual trajectory. There may be biases in the target planet ephemerides and the gravitational constants of the Sun and target planet. The numerical accuracy and the number of celestial bodies employed in the generation of the actual trajectory may also differ.

Ephemeris biases are specified as biases in orbital elements a, e, i, Ω, ω, and M.. However, within the program are stored the ephemeris constants of a, e, i, Ω, $\tilde{\omega}$, and M for the planets and a, e, i, Ω, $\tilde{\omega}$, and L for the moon, where

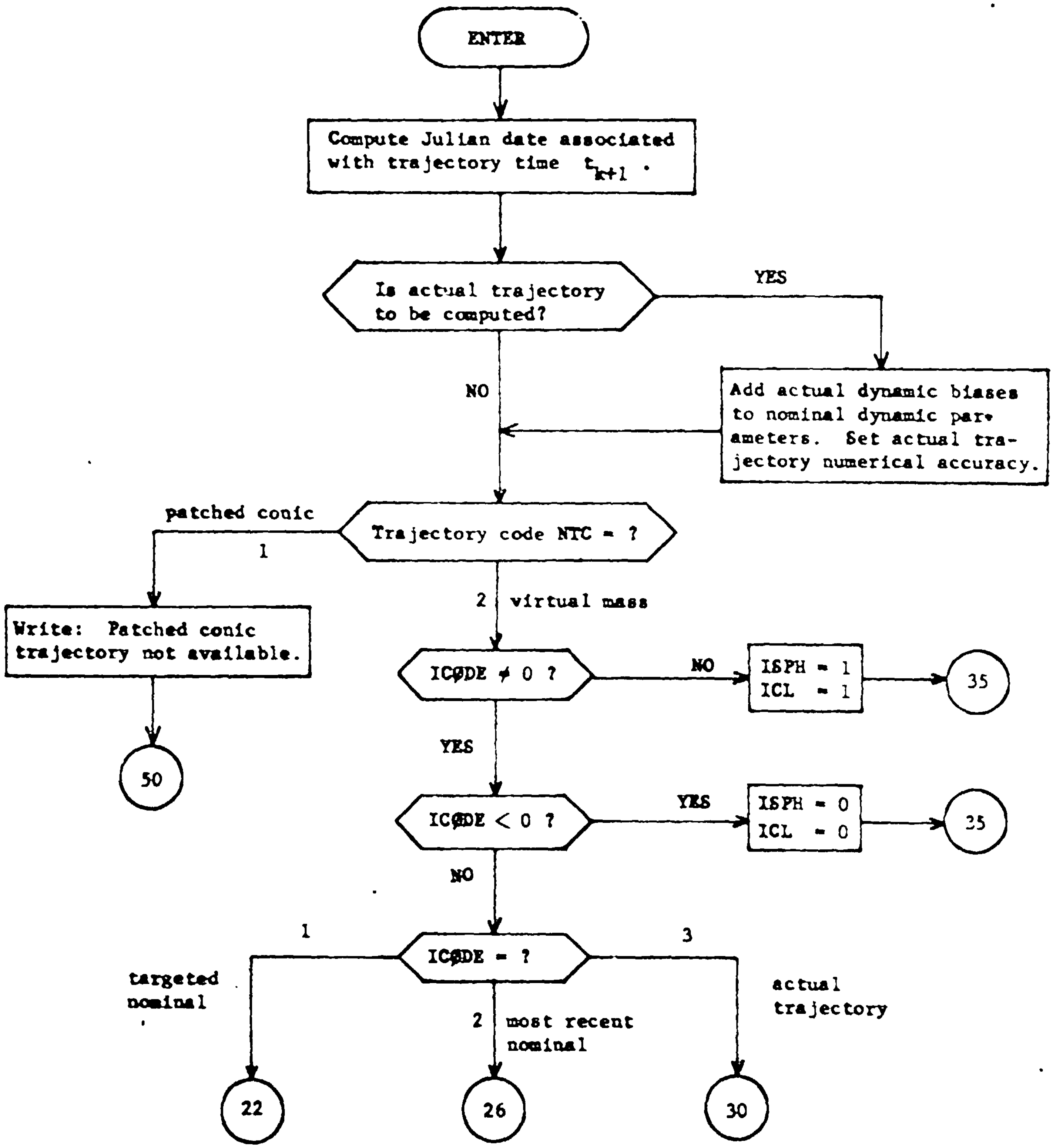
$$\tilde{\omega} = \omega + \Omega$$

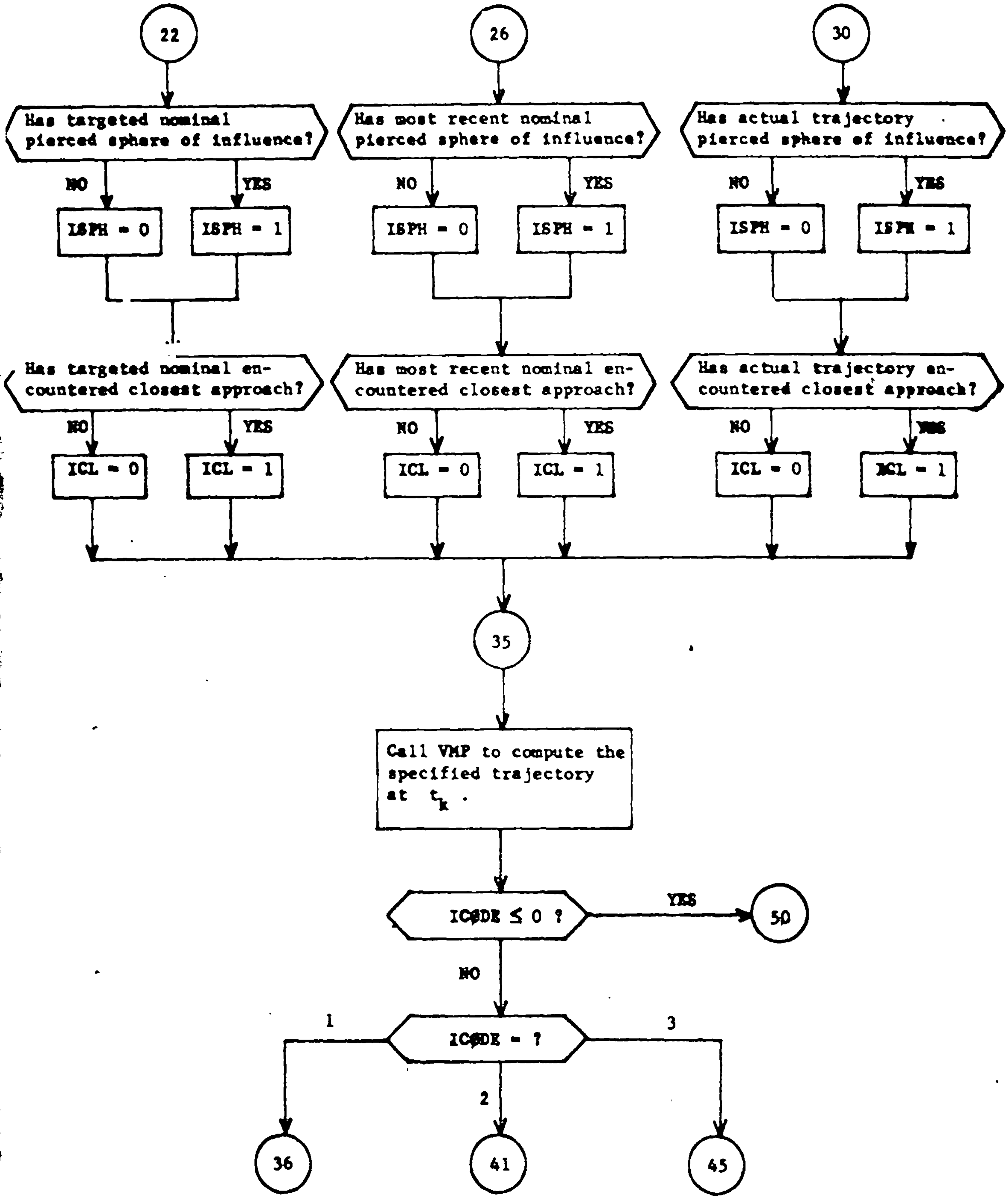
and

$$L = M + \omega + \Omega .$$

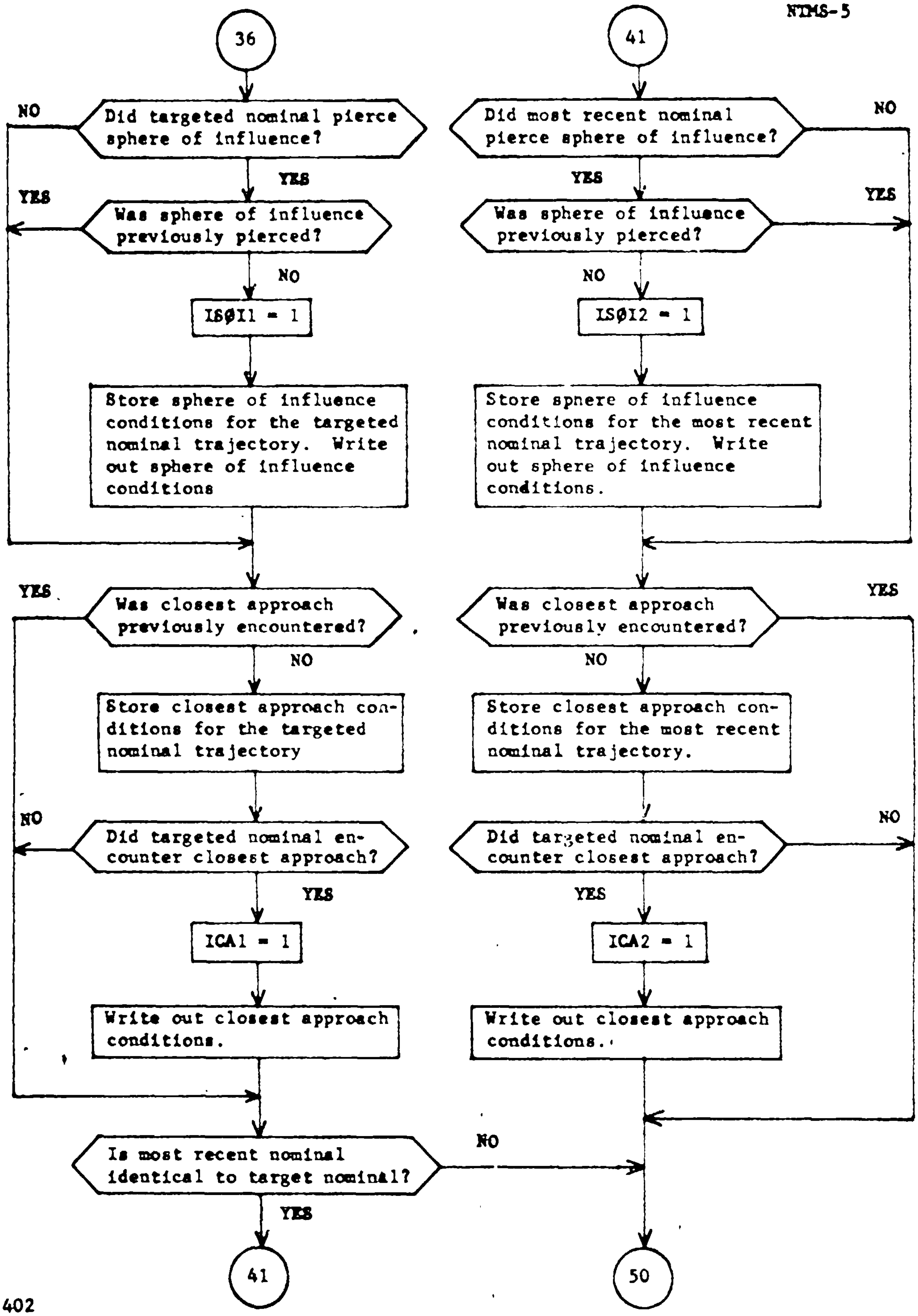
Incrementation of $\tilde{\omega}$ and L requires addition of biases in Ω, ω, and M as indicated by the above equations.

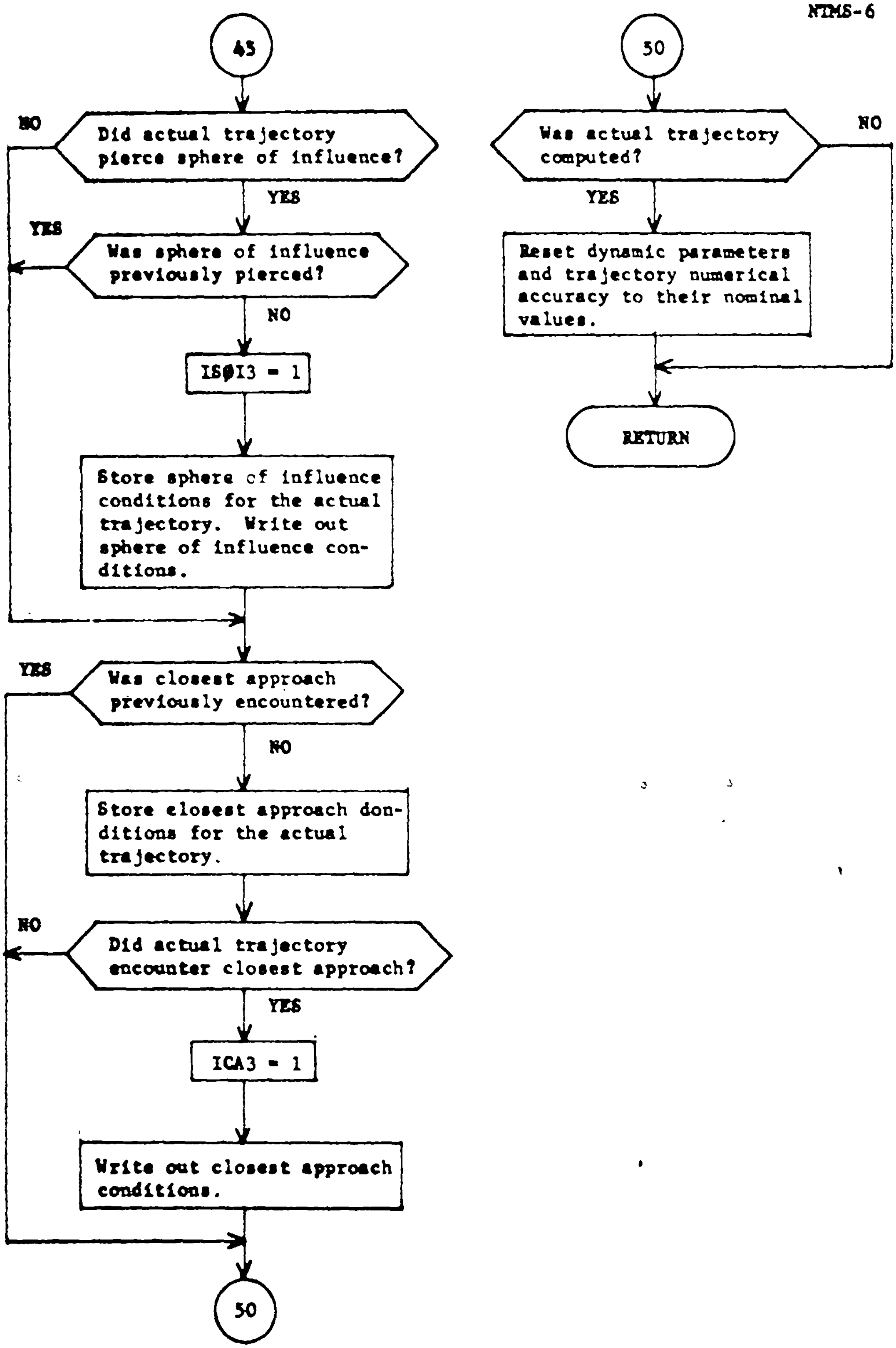
NTMS Flow Chart





22
26
30
Has targeted nominal pierced sphere of influence?
Has most recent nominal pierced sphere of influence?
Has actual trajectory pierced sphere of influence?
NO
YES
ISPH = 0
ISPH = 1
Has targeted nominal encountered closest approach?
Has most recent nominal encountered closest approach?
Has actual trajectory encountered closest approach?
ICL = 0
ICL = 1
35
Call VMP to compute the specified trajectory at t_k .
ICØDE ≤ 0 ?
50
ICØDE = ?
1
2
3
36
41
45


36
41
Did targeted nominal pierce sphere of influence?
NO
YES
Was sphere of influence previously pierced?
YES
NO
ISØI1 = 1
Store sphere of influence conditions for the targeted nominal trajectory. Write out sphere of influence conditions
Was closest approach previously encountered?
YES
NO
Store closest approach conditions for the targeted nominal trajectory
Did targeted nominal encounter closest approach?
NO
YES
ICA1 = 1
Write out closest approach conditions.
Is most recent nominal identical to target nominal?
NO
YES
41
Did most recent nominal pierce sphere of influence?
NO
YES
Was sphere of influence previously pierced?
YES
NO
ISØI2 = 1
Store sphere of influence conditions for the most recent nominal trajectory. Write out sphere of influence conditions.
Was closest approach previously encountered?
YES
NO
Store closest approach conditions for the most recent nominal trajectory.
Did targeted nominal encounter closest approach?
NO
YES
ICA2 = 1
Write out closest approach conditions.
50


45
Did actual trajectory pierce sphere of influence?
NO
YES
Was sphere of influence previously pierced?
YES
NO
ISØI3 = 1
Store sphere of influence conditions for the actual trajectory. Write out sphere of influence conditions.
Was closest approach previously encountered?
YES
NO
Store closest approach donditions for the actual trajectory.
Did actual trajectory encounter closest approach?
NO
YES
ICA3 = 1
Write out closest approach conditions.
50
50
Was actual trajectory computed?
NO
YES
Reset dynamic parameters and trajectory numerical accuracy to their nominal values.
RETURN

SUBROUTINE ORB

PURPOSE: TO COMPUTE THE ORBITAL ELEMENTS -- INCLINATION, LONGITUDE OF ASCENDING NODE, LONGITUDE OF PERIHELION, ECCENTRICITY, AND LENGTH OF SEMIMAJOR AXIS -- FOR A SPECIFIED PLANET AT A GIVEN TIME.

CALLING SEQUENCE: CALL ORB(IP,D)

ARGUMENT: D I JULIAN DATE, EPOCH 1900, OF THE TIME AT WHICH THE ELEMENTS ARE TO BE CALCULATED

IP I CODE NUMBER OF PLANET
=1 SUN
=2 MERCURY
=3 VENUS
=4 EARTH
=5 MARS
=6 JUPITER
=7 SATURN
=8 URANUS
=10 PLUTO
=11 MOON

SUBROUTINES SUPPORTED: DATA DATAS PCTM PRINT3 PRINT4 PSIM TRAKM TRAKS TRAPAR VMP GUIDM GUID GUISIM GUISS PRNTS3 HELIO LAUNCH LUNTAR MULCON MULTAR PRNTS4 TRAPAR

SUBROUTINES REQUIRED: NONE

LOCAL SYMBOLS: FN1 STATEMENT FUNCTION DEFINING A THIRD ORDER POLYNOMIAL

FN2 STATEMENT FUNCTION DEFINING A FIRST ORDER POLYNOMIAL

ITEMP INTERMEDIATE VARIABLE

PI2 TWICE THE MATHEMATICAL CONSTANT PI

COMMON COMPUTED/USED: ELMNT T

COMMON USED: CN EMN SMJR ST TWOPI

ORB Analysis

ORB determines the mean orbital elements for any gravitational body at a specified time.

The elements used are semi-major axis a , eccentricity e , inclination i , longitude of the ascending node Ω , and longitude of periapsis $\tilde{\omega}$. These elements are referenced to heliocentric ecliptic for the planets or geocentric ecliptic for the moon.

The mean elements are computed from time expansions as follows. Let α be any of the elements. Then the value of α at any time t is given by

$$\alpha(t) = \alpha_o + \alpha_1 t + \alpha_2 t^2 + \alpha_3 t^3$$

where the constants α_k are stored by BLKDAT. These constants are stored into the arrays CN, ST, and EMN for inner planets, outer planets, and the moon respectively. The definitions of these arrays and the values stored are provided in the analysis of the previous subroutine BLKDAT. The element value as computed from the above equation is then returned in the ELMNT array according to the gravitational body code k as

ELMNT(8k-15) = i	k = 1 Sun	= 1 Saturn
ELMNT(8k-14) = Ω	2 Mercury	8 Uranus
ELMNT(8k-13) = $\tilde{\omega}$	3 Venus	9 Neptune
ELMNT(8k-12) = e	4 Earth	10 Pluto
ELMNT(8k-10) = a	5 Mars	11 Moon
ELMNT(8k-9) = ω	6 Jupiter	

SUBROUTINE PARTL

PURPOSE COMPUTE PARTIALS OF B DOT T AND B DOT R WITH RESPECT TO SPACECRAFT POSITION AND VELOCITY

CALLING SEQUENCE: CALL PARTL(R,V,B,BDT,BDR,PBT,PBR)

ARGUMENT:	B	O	IMPACT PLANE PARAMETER
	BDR	O	B DOT R
	BDT	O	B DOT T
	PBR	O	PARTIAL OF B DOT R WITH RESPECT TO R AND V
	PBT	O	PARTIAL OF B DOT T WITH RESPECT TO R AND V
	R	I	POSITION OF VEHICLE RELATIVE TO PLANET
	V	I	VELOCITY OF VEHICLE RELATIVE TO PLANET

SUBROUTINES SUPPORTED: GUISS GUID

LOCAL SYMBOLS:	H3	INTERMEDIATE VARIABLE
	RU	INTERMEDIATE VARIABLE
	S	MAGNITUDE OF VELOCITY
	U	INTERMEDIATE VARIABLE
	U2	SQUARE OF U
	U2PV2	INTERMEDIATE VARIABLE
	UV	INTERMEDIATE VARIABLE
	UV3	CUBE OF UV
	V2	SQUARE OF MAGNITUDE OF VELOCITY

COMMON USED: ZERO

PARTL Analysis

PARTL is responsible for the computation of the partials of $B \cdot T$ and $B \cdot R$ with respect to the cartesian components of position and velocity.

Let the state of the spacecraft with respect to the target body at intersection with its sphere of influence be denoted

$$\vec{r} = \left[x,\ y,\ z\right]^T \qquad r = \sqrt{x^2 + y^2 + z^2} \tag{1}$$

$$\vec{v} = \left[\dot{x},\ \dot{y},\ \dot{z}\right]^T \qquad v = \sqrt{\dot{x}^2 + \dot{y}^2 + \dot{z}^2} \tag{2}$$

Introduce the approach asymptote $\hat{S}$ and approximate it by the direction of $\vec{v}$.

$$\hat{S} = \frac{\vec{v}}{v} \tag{3}$$

The B-plane is the plane normal to $\hat{X}$ containing the center of the target body. Any vector $\vec{\beta}$ within the B-plane must satisfy therefore

$$\hat{S} \cdot \vec{\beta} = 0 \tag{4}$$

The impact parameter vector $\vec{B}$ is determined by the intersection of the B-plane and the incoming asymptote. The incoming asymptote is given parametrically by

$$\vec{\gamma} = \vec{r} + \vec{v}\, t \tag{5}$$

The time at which the asymptote intersects the B-plane may be determined by applying the B-plane condition (4)

$$\hat{S} \cdot \vec{r} + \hat{S} \cdot \vec{v}\, t = 0$$

$$t = -\frac{\vec{r} \cdot \vec{v}}{v^2} \tag{6}$$

Therefore the B-vector is given by

$$\vec{B} = \vec{r} - \frac{\vec{r} \cdot \vec{v}}{v^2}\, \vec{v}$$

$$\vec{B} = \left[x - \alpha\dot{x},\ y - \alpha\dot{y},\ z - \alpha\dot{z}\right]^T \tag{7}$$

where $\quad \alpha = \dfrac{\vec{r} \cdot \vec{v}}{v^2} = \dfrac{x\dot{x} + y\dot{y} + z\dot{z}}{\dot{x}^2 + \dot{y}^2 + \dot{z}^2}$

Now assuming that the T axis is to lie in the x-y reference plane and the B-plane, it is defined as

$$\hat{T} = \frac{\hat{S} \times \hat{K}}{|\hat{S} \times \hat{K}|}$$

$$\hat{T} = \left[\frac{\dot{y}}{u} , - \frac{\dot{x}}{u} , 0 \right] \tag{8}$$

where $u^2 = \dot{x}^2 + \dot{y}^2$. The $\hat{R}$ axis is defined by

$$\hat{R} = \hat{S} \times \hat{T}$$

$$\hat{R} = \frac{1}{uv} \left[\dot{x}\dot{z}, \ \dot{y}\dot{z}, \ -u^2 \right]^T \tag{9}$$

Now combining (7), (8), (9) B·T and B·R may be computed in terms of the state components

$$B \cdot T = \frac{1}{u} (x\dot{y} - \dot{x}y)$$

$$B \cdot R = \frac{1}{uv} \left[(x\dot{x} + y\dot{y})\dot{z} - u^2 z \right] \tag{10}$$

where $u^2 = \dot{x}^2 + \dot{y}^2$, $v^2 = u^2 + \dot{z}^2$.

The partials may now be computed from differentiation of the above equations.

$$\frac{\partial B \cdot T}{\partial x} = \frac{\dot{y}}{u} \qquad \frac{\partial B \cdot R}{\partial x} = \frac{\dot{x}\dot{z}}{uv}$$

$$\frac{\partial B \cdot T}{\partial v} = - \frac{\dot{x}}{u} \qquad \frac{\partial B \cdot R}{\partial y} = \frac{\dot{y}\dot{z}}{uv}$$

$$\frac{\partial B \cdot T}{\partial z} = 0 \qquad \frac{\partial B \cdot R}{\partial z} = - \frac{u}{v}$$

$$\frac{\partial B \cdot T}{\partial x} = - \frac{\dot{y}}{u^3} (x\dot{x} + y\dot{y}) \qquad \frac{\partial B \cdot R}{\partial \dot{x}} = \frac{\dot{z}}{u^3 v^3} \left[u^2 (v^2 x - \dot{x} z \dot{z}) - \dot{x}(u^2 + v^2)(x\dot{x} + y\dot{y}) \right]$$

$$\frac{\partial B \cdot T}{\partial y} = \frac{\dot{x}}{u^3} (x\dot{x} + y\dot{y}) \qquad \frac{\partial B \cdot R}{\partial \dot{y}} = \frac{\dot{z}}{u^3 v^3} \left[u^2 (v^2 y - \dot{y} z \dot{z}) - \dot{y}(u^2 + v^2)(x\dot{x} + y\dot{y}) \right]$$

$$\frac{\partial B \cdot T}{\partial z} = 0 \qquad \frac{\partial B \cdot R}{\partial z} = \frac{u}{v^3} (x\dot{x} + y\dot{y} + z\dot{z})$$

SUBROUTINE PCTM

PURPOSE: CONTROL COMPUTATION OF STATE TRANSITION MATRIX USING THE ANALYTICAL PATCHED CONIC TECHNIQUE

CALLING SEQUENCE: CALL PCTM(RI)

ARGUMENT: RI I POSITION AND VELOCITY OF THE VEHICLE AT THE BEGINNING OF THE TIME INTERVAL

SUBROUTINES SUPPORTED: PSIM

SUBROUTINES REQUIRED: CONC2 EPHEM ORB

LOCAL SYMBOLS:

D	JULIAN DATE, EPOCH JAN.0, 1900, OF INITIAL TIME
DELT	LENGTH OF TIME INCREMENT IN PROPER UNITS
DUM	TEMPORARY STORAGE FOR STATE TRANSITION MATRIX
GMS	GRAVITATIONAL CONSTANT OF GOVERNING BODY
IP	CODE OF PLANET
RM	DISTANCE FROM SPECIFIED PLANET
RS	POSITION OF VEHICLE RELATIVE TO SPECIFIED PLANET
VS	VELOCITY OF VEHICLE RELATIVE TO SPECIFIED PLANET

COMMON COMPUTED/USED: XP

COMMON COMPUTED: NO PHI

COMMON USED: ALNGTH DATEJ DELTM F IBARY NBOD NB PMASS SPHERE TM TRTM1

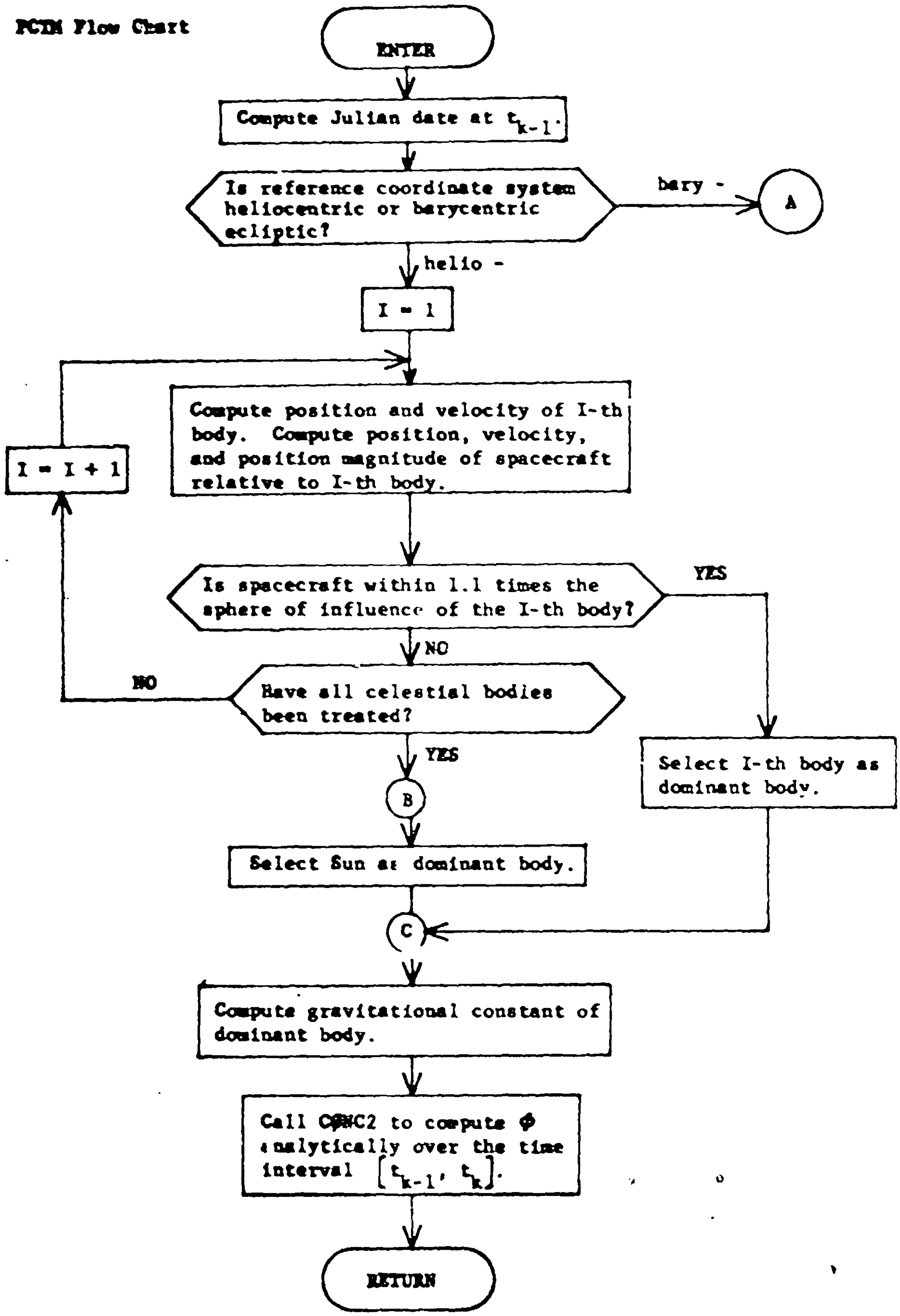
PCTM Analysis

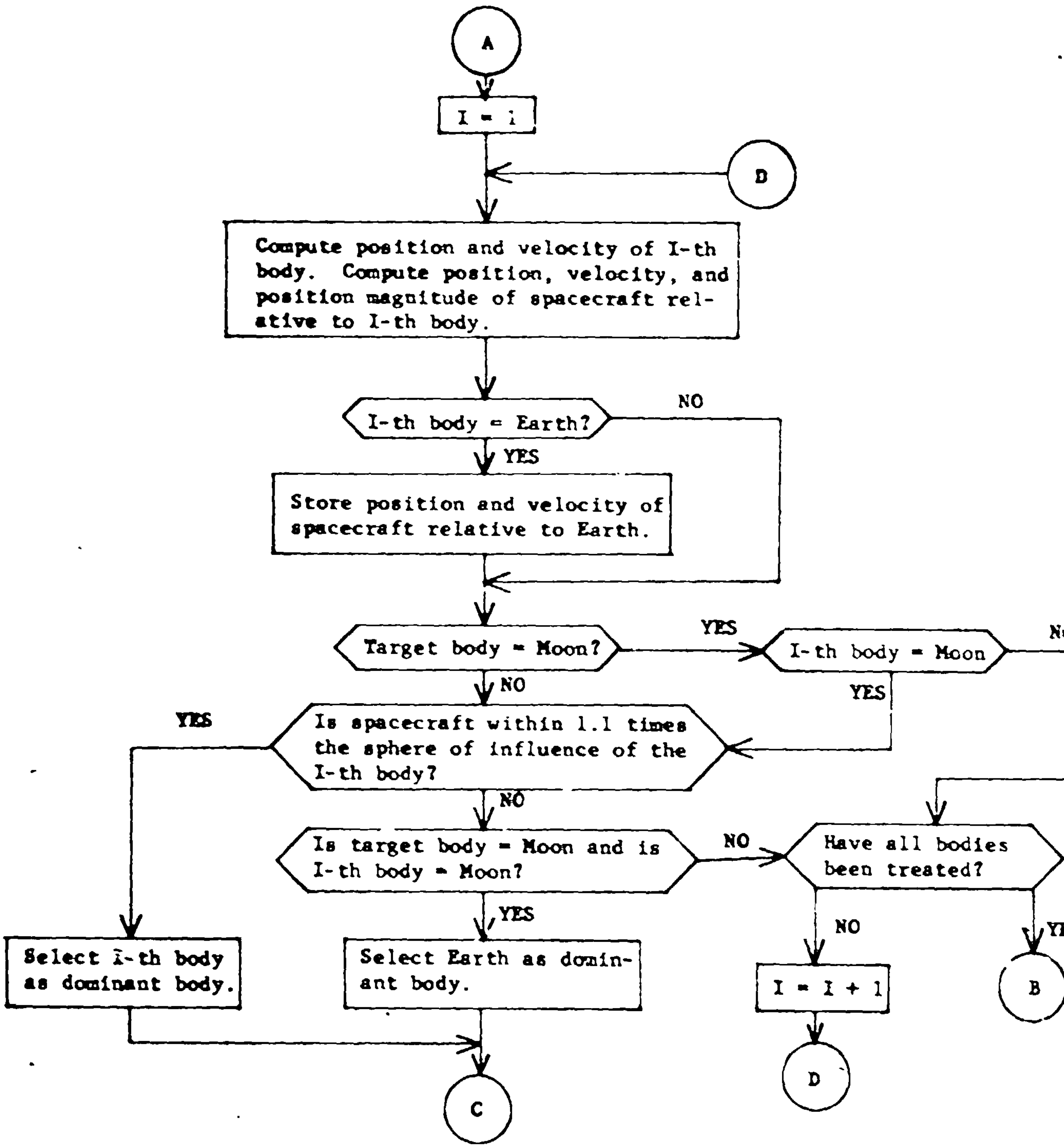
Subroutine PCTM does not actually compute the state transition matrix $\Phi(t_k, t_{k-1})$ itself; this is accomplished by calling CØNC2 from within PCTM. The primary function of PCTM is to determine the dominant body at time t_{k-1} to be used in the computation of $\Phi(t_k, t_{k-1})$ by means of the analytical patched conic technique.

On interplanetary trajectories we compute the distance separating the spacecraft from each of the celestial bodies included in the analysis. If the distance between the spacecraft and the i-th body is less than or equal to 1.1 times the sphere of influence of the i-th body, the i-th body is selected as the dominant body. Otherwise, the Sun is selected as the dominant body.

On luna trajectories we compute the distance separating the spacecraft from the Moon. If this distance is less than or equal to 1.1 times the sphere of influence of the Moon, the moon is selected as the dominant body. If not, the Earth is selected as the dominant body.

PCTM Flow Chart




A
I = 1
D
Compute position and velocity of I-th body. Compute position, velocity, and position magnitude of spacecraft relative to I-th body.
I-th body = Earth?
NO
YES
Store position and velocity of spacecraft relative to Earth.
Target body = Moon?
YES
I-th body = Moon
NO
NO
YES
YES
Is spacecraft within 1.1 times the sphere of influence of the I-th body?
NO
Is target body = Moon and is I-th body = Moon?
NO
Have all bodies been treated?
YES
NO
YES
Select I-th body as dominant body.
Select Earth as dominant body.
I = I + 1
B
D
C

SUBROUTINE PECEQ

PURPOSE: TO COMPUTE THE MATRIX DEFINING THE TRANSFORMATION FROM PLANET CENTERED ECLIPTIC COORDINATES TO PLANET CENTERED EQUATORIAL COORDINATES AS A FUNCTION OF THE PARTICULAR PLANET AND TIME.

CALLING SEQUENCE: CALL PECEQ(NP,D,ECEQ)

ARGUMENT	NP	I	CODE OF PLANET
	D	I	JULIAN DATE, EPOCH 1900, OF REFERENCE TIME
	ECEQ(3,3)	O	TRANSFORMATION MATRIX FROM ECLIPTIC TO EQUATORIAL COORDINATES

SUBROUTINES SUPPORTED: TARGET HELIO LAUNCH LUNTAR MULTAR INSERS TRAPAR VMP DATAS GUISIM DATA GUIDM EXCUTE

SUBROUTINES REQUIRED: EULMX

LOCAL SYMBOLS:

DD	JULIAN TIME (EPOCH 1900) DIVIDED BY 10000.
DGTR	CONVERSION FACTOR FROM DEGREES TO RADIANS
ECOP	TRANSFORMATION MATRIX FROM ECLIPTIC TO ORBITAL PLANE
OPEQ	TRANSFORMATION MATRIX FROM ORBITAL TO EQUATORIAL PLANE
T	YEARS OF JULIAN TIME (EPOCH 1900)
XI	THE INCLINATION OF THE ORBITAL PLANE TO ECLIPTIC PLANE
XIQ	INCLINATION OF PLANET EQUATOR TO ORBITAL PLANE
XL	THE LONGITUDE OF THE ASCENDING NODE OF THE ORBITAL PLANE TO THE ECLIPTIC PLANE
XLQ	THE LONGITUDE OF THE ASCENDING NODE OF THE PLANET EQUATOR TO THE ORBITAL PLANE

COMMON USED: EMN EM2 EM4 EM5 EM6 EM7 EM8 ZERO

PECEQ Analysis

PECEQ is responsible for the computation of the transformation matrix from ecliptic to planet equatorial coordinates for any planet (or moon) at any time.

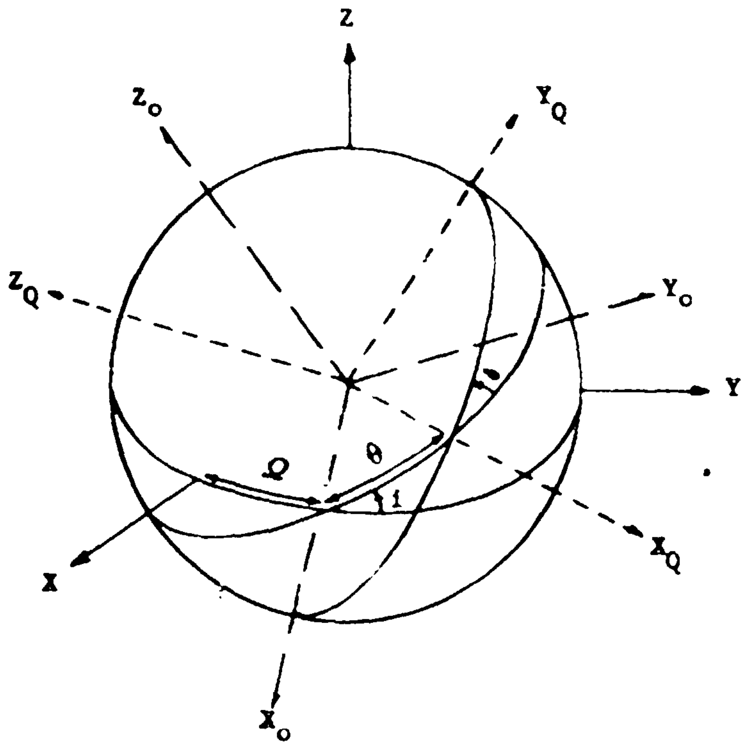


Figure 1. Geometry of Ecliptic, Orbital, and Equatorial Planes

Figure 1 illustrates the geometry of the problem. In that figure the following definitions hold

XYZ	The ecliptic coordinate axes
$X_oY_oZ_o$	The orbital plane coordinate axes
$X_QY_QZ_Q$	The planet equatorial coordinate axes
i	Inclination of orbital plane to ecliptic plane
Ω	Right ascension of orbital plane to ecliptic plane
δ	Inclination of planet equator to orbital plane
θ	Right ascension of planet equatorial to orbital plane

Time expansions for the mean values of the angles are evaluated at the specific time. Then the construction of the transformation matrix proceeds in two steps:

In the first step the transformation from the ecliptic to orbital plane coordinates is made. This is done by rotating about the z-axis through an angle Ω and then about the resulting x-axis through an angle i :

$$T_{ECOP} = (\ \Omega \text{ about } 3,\ i \text{ about } 1)$$

The transformation from the orbital plane to the equatorial coordinates is accomplished in a similar fashion:

$$T_{ECOP} = (\ \theta \text{ about } -3,\ \delta \text{ about } -1)$$

The rotation matrix (or Euler matrix) for each of these transformations is computed by subroutine EULMX. The transformation from ecliptic to equatorial is now given as the product of these matrices

$$T_{ECEQ} = T_{OPEQ}\ T_{ECOP}$$

SUBROUTINE PERHEL

PURPOSE: TO PROPAGATE A HELIOCENTRIC TRAJECTORY CONSIDERING THE PERTUBATIONS PRODUCED BY BOTH THE LAUNCH AND TARGET BODIES.

CALLING SEQUENCE: CALL PERHEL(GM,HSI,HLTI,HLTF,DELT,HSF)

ARGUMENT:

GM(3)	I	GRAVITATIONAL CONSTANTS OF SUN,LAUNCH AND TARGET PLANETS
HSI(6)	I	HELIOCENTRIC ECLIPTIC SPACECRAFT STATE (INITIAL)
HLTI(2,3)	I	INITIAL HELIOCENTRIC STATES OF LAUNCH AND TARGET BODIES
HLTF(2,3)	I	FINAL HELIOCENTRIC STATES OF LAUNCH AND TARGET BODIES
DELT	I	TIME INTERVAL OF PROPAGATION
HSF(6)	O	HELIOCENTRIC ECLIPTIC SPACECRAFT STATE (FINAL)

SUBROUTINES SUPPORTED: PULCOV PULSEX

SUBROUTINES REQUIRED: BATCON

LOCAL SYMBOLS:

CON	INTERMEDIATE VARIABLE
DELR	RF-RI
PER	PERTURBATION IN FINAL STATE
PSF	SPACECRAFT POSITION RELATIVE TO PLANET (FINAL)
PSI	SPACECRAFT POSITION RELATIVE TO PLANET (INITIAL)
RAV	AVERAGE OF RI AND RF
RA	INTERMEDIATE VARIABLE
RF	MAGNITUDE OF PSF
RH	INTERMEDIATE VARIABLE
RI	MAGNITUDE OF PSI

PERHEL Analysis

PERHEL is responsible for propagating a heliocentric trajectory considering the perturbations produced by both the launch and target bodies. The equations of motion of a body moving under the influence of the sun while perturbed by a smaller mass are

$$\ddot{\vec{r}} = -\frac{\mu_o \vec{r}}{r^3} - \frac{\mu(\vec{r} - \vec{r}_m)}{\left|\vec{r} - \vec{r}_m\right|^3} - \frac{\mu \vec{r}_m}{r_m{}^3} \qquad (1)$$

where $\vec{r}$ is the vector radius from the sun to the spacecraft
$\vec{r}_m$ is the vector radius from the sun to the perturbative mass
μ_o, μ are the gravitational constants of the sun and mass respectively.

Assuming that the indirect term is small, attention may be directed to the first two terms only. Suppose that $(\vec{r}_o(t), \vec{v}_o(t))$ satisfy

$$\begin{aligned} \dot{\vec{r}}_o &= \vec{v}_o \\ \dot{\vec{v}}_o &= \frac{\mu_o \vec{r}_o}{r_o{}^3} \end{aligned} \qquad (2)$$

Then $(r_o(t), v_o(t))$ are given by the familar equations of conic motion.

A first order corrected solution necessary to account for the direct term force must then satisfy

$$\begin{aligned} \dot{\vec{r}} &= \dot{\vec{r}}_o + \dot{\vec{\delta r}} = \vec{v} \\ \dot{\vec{v}} &= \dot{\vec{v}} + \dot{\vec{\delta v}} = -\frac{\mu_o \vec{r}_o}{r_o{}^3} - \mu \frac{(\vec{r}_2 - \vec{r}_m)}{\left|\vec{r}_o - \vec{r}_m\right|^3} \end{aligned} \qquad (3)$$

Applying the conditions (2) leads to the equations defining the corrections

$$\begin{aligned} \dot{\vec{\delta r}} &= \vec{\delta v} \\ \dot{\vec{\delta v}} &= -\mu \frac{\vec{R}}{R^3} \end{aligned} \qquad (4)$$

where $\vec{R} = \vec{r}_o(t) - \vec{r}_m(t)$ is the position vector of the spacecraft with respect to the perturbing mass.

One further assumption enables one to solve in closed form the perturbations produced by the third mass. Generally $\vec{R}\,(t)$ and $R\,(t)$ are nearly linear functions of time. Therefore suppose that the initial and final values of these variables are known to be $\vec{R}_1$, $\vec{R}_2$, R_1, R_2 over the interval Δt.

Introduce the definitions

$$\overrightarrow{\Delta R} = \vec{R}_2 - \vec{R}_1$$
$$\Delta R = R_2 - R_1 \quad (\text{not } |\overrightarrow{\Delta R}|)$$
$$\langle R \rangle = \tfrac{1}{2}(R_1 + R_2) \tag{5}$$
$$\hat{\Delta R} = \frac{\vec{R}_2}{R_2} - \frac{\vec{R}_1}{R_1}$$

Then the equation defining the velocity perturbation would be

$$\vec{\delta\dot{v}} = -\mu \frac{\vec{a} + \vec{b}\,t}{(c + d\,t)^3} \qquad \vec{a} = \vec{R}_1 \qquad c = R_1$$
$$\vec{b} = \frac{\overrightarrow{\Delta R}}{\Delta t} \qquad d = \frac{\Delta R}{\Delta t} \tag{6}$$

It is more convenient however to transform from time t to position magnitude ρ as the independent variable. This may be done since the position magnitude is assumed to be linear in time with $\dot{\rho} = \frac{\Delta R}{\Delta t}$.
According to the assumptions, the position vector $\vec{R}$ is a linear function of ρ also

$$\vec{R} = \vec{A} + \vec{B}\,\rho \tag{7}$$

Since $\vec{R}(\rho_1) = \vec{R}_1$ and $\vec{R}(\rho_2) = \vec{R}_2$, the constants are

$$\vec{A} = \vec{R}_1 - \frac{\overrightarrow{\Delta R}}{\Delta R} R_1 = -\frac{R_1 R_2}{\Delta R}\hat{\Delta R}$$
$$\vec{B} = \frac{\overrightarrow{\Delta R}}{\Delta R} \tag{8}$$

In terms of ρ the equations defining the perturbations may be written (with primes indicating differentiation with respect to ρ)

$$\overrightarrow{\delta r'} = \frac{\Delta t}{\Delta R}\overrightarrow{\delta v}$$
$$\overrightarrow{\delta v'} = -\frac{\mu \Delta t}{\Delta R}\frac{\vec{A} + \vec{B}\rho}{\rho^3} \tag{9}$$

These equations are easily integrated to determine the perturbations caused as the spacecraft moves from $\vec{R}_1$ to $\vec{R}_2$ relative to the perturbative body:

$$\vec{\delta v} = -\frac{\mu\Delta t}{\Delta R}\int_{R_1}^{\rho}\frac{\vec{A}+\vec{B}\rho}{\rho^3}\,d\rho$$

$$= \frac{\mu\Delta t}{\Delta R}\left[\frac{\vec{A}}{2}\left(\frac{1}{\rho^2}-\frac{1}{R_1^{\,2}}\right)+\vec{B}\left(\frac{1}{\rho}-\frac{1}{R_1}\right)\right]$$

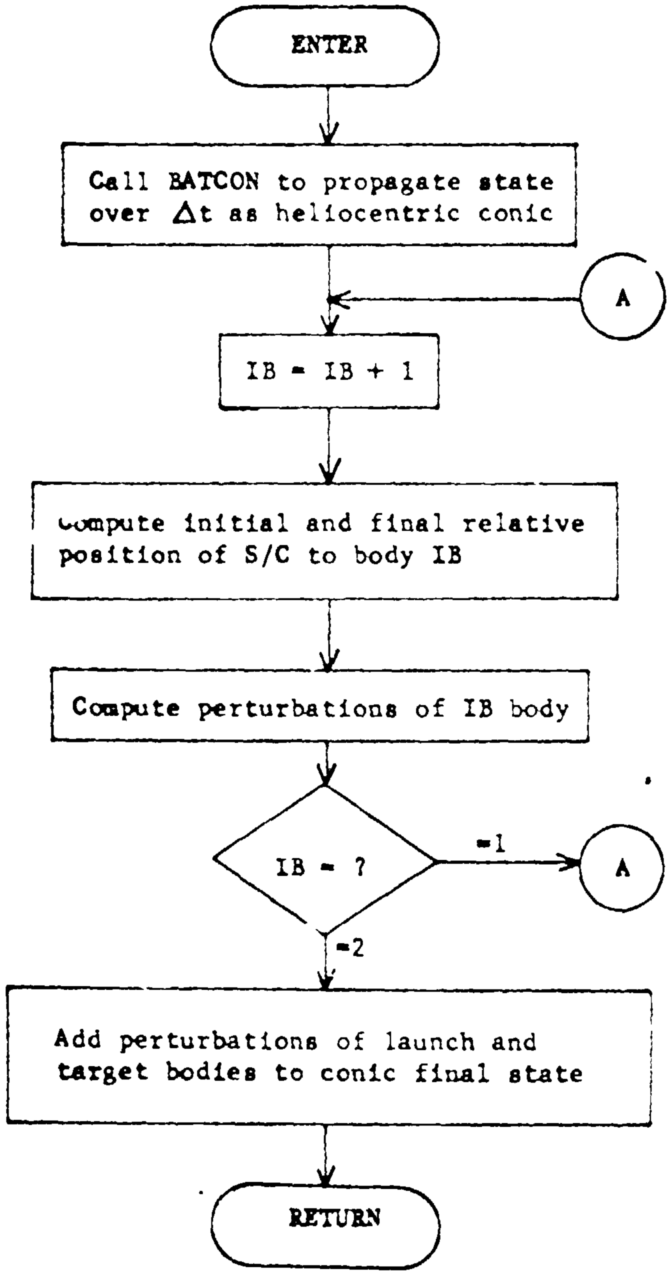
$$= \frac{\mu\Delta t}{R_1R_2\Delta R}\left[\langle R\rangle\,\hat{\Delta R}-\vec{\Delta R}\right], \qquad \rho = R_2 \tag{10}$$

$$\vec{\delta r} = \frac{\mu\Delta t^2}{\Delta R^2}\int_{R_1}^{R_2}\left[\frac{\vec{A}}{2}\left(\frac{1}{\rho^2}-\frac{1}{R_1^{\,2}}\right)+\vec{B}\left(\frac{1}{\rho}-\frac{1}{R_1}\right)\right]d\rho$$

$$= \frac{\mu\Delta t^2}{\Delta R}\left[\frac{1}{2}\,\frac{\hat{\Delta R}}{R_1}+\frac{\vec{\Delta R}}{\Delta R^2}\left(\ln\left(\frac{R_2}{R_1}\right)-\frac{\Delta R}{R_1}\right)\right] \tag{11}$$

PERHEL calls BATCON for the generation of the uncorrected heliocentric conic, computes the initial and final positions of the spacecraft relative to each of the launch and target planets, and computes the perturbations based on equations (10) and (11) above.

PERHEL Flow Chart



SUBROUTINE PLND

PURPOSE: TO COMPUTE COLUMNS OF THE STATE TRANSITION MATRIX PARTITIONS TXXS AND TXU ASSOCIATED WITH TARGET PLANET EPHEMERIS BIASES INCLUDED IN THE AUGMENTED STATE VECTOR BY A NUMERICAL DIFFERENCING TECHNIQUE.

CALLING SEQUENCE: CALL PLND(RI,RF)

ARGUMENT: RF I POSITION AND VELOCITY OF THE VEHICLE AT THE END OF THE TIME INTERVAL

RI I POSITION AND VELOCITY OF THE VEHICLE AT THE BEGINNING OF THE TIME INTERVAL

SUBROUTINES SUPPORTED: PSIM

SUBROUTINES REQUIRED: NTM

LOCAL SYMBOLS:

DEL	TEMPORARY STORAGE FOR TARGET PLANET SEMI-MAJOR AXIS FACTOR USED IN NUMERICAL DIFFERENCING
IC	COUNTER FOR VARIABLES AUGMENTED TO STATE VECTOR
IEMN	VECTOR OF INDICES FOR ORBITAL ELEMENTS OF THE MOON
IEND	FLAG FOR VARIABLES AUGMENTED TO STATE VECTOR
IPR	TEMPORARY STORAGE FOR IPRINT
NEW	VECTOR OF INDICES FOR ORBITAL ELEMENTS OF INNER AND OUTER PLANETS
RPER	ALTERED FINAL POSITION AND VELOCITY OF VEHICLE
SAVE1	TEMPORARY STORAGE FOR CONSTANTS OF AUGUMENTED ELEMENTS OF TARGET PLANET
SAVE2	SAME COMMENTS AS SAVE1

COMMON COMPUTED/USED: CN EMN IPRINT SMJR ST

COMMON COMPUTED: TXU TXXS

COMMON USED: ALNGTH DELAXS DELX IAUGDC IAUGIN IAUG NTMC NTP

PLND Analysis

The nonlinear equations of motion of the spacecraft can be written symbolically as

$$\dot{\vec{x}} = \vec{f}(\vec{x}, \vec{e}(t), t) \qquad (1)$$

where $\vec{x}$ is the spacecraft position/velocity state and $\vec{e}(t)$ is a vector composed of the 6 orbital elements a, e, i, Ω, ω, and M of the target planet. The motion of the spacecraft is, of course, dependent on the positions of other celestial bodies, but this dependency need not be explicitly stated for the purposes of this analysis.

Suppose we wish to use numerical differencing to compute those columns of θ_{xx_s} and θ_{xu} associated with target planet ephemeris biases included in the augmented state vector over the time interval $[t_{k-1}, t_k]$. Let $\vec{\theta}_j(t_k, t_{k-1})$ represent the column associated with the j-th ephemeris bias. We assume we have available the nominal states $\vec{x}^*(t_{k-1})$ and $\vec{x}^*(t_k)$, which, of course, were obtained by numerically solving equation (1) using nominal $\vec{e}(t)$. To obtain $\vec{\theta}_j(t_k, t_{k-1})$ we increment the j-th orbital element by the pertinent numerical differencing factor Δe_j and numerically integrate equation (1) over the interval $[t_{k-1}, t_k]$ to obtain the new spacecraft state $\vec{x}_j(t_k)$, where the j-subscript on the spacecraft state indicates that it was obtained by incrementing the j-th orbital element. Then

$$\vec{\theta}_j(t_k, t_{k-1}) = \frac{\vec{x}_j(t_k) - \vec{x}^*(t_k)}{\Delta e_j} . \qquad (2)$$

Ephemeris biases are defined as biases of the basic set of orbital elements a, e, i, Ω, ω, and M. However, within the program are stored the ephemeris constants of a, e, i, Ω, $\tilde{\omega}$, and M for the planets and a, e, i, Ω, $\tilde{\omega}$, and L for the moon. Thus, in order to increment certain of the basic elements we must increment certain combinations of the stored ephemeris constants.

The elements ω and M are related to the longitude of perihelion $\tilde{\omega}$ and the mean longitude L as follows:

$$\omega = \tilde{\omega} - \Omega$$

$$M = L - \tilde{\omega}$$

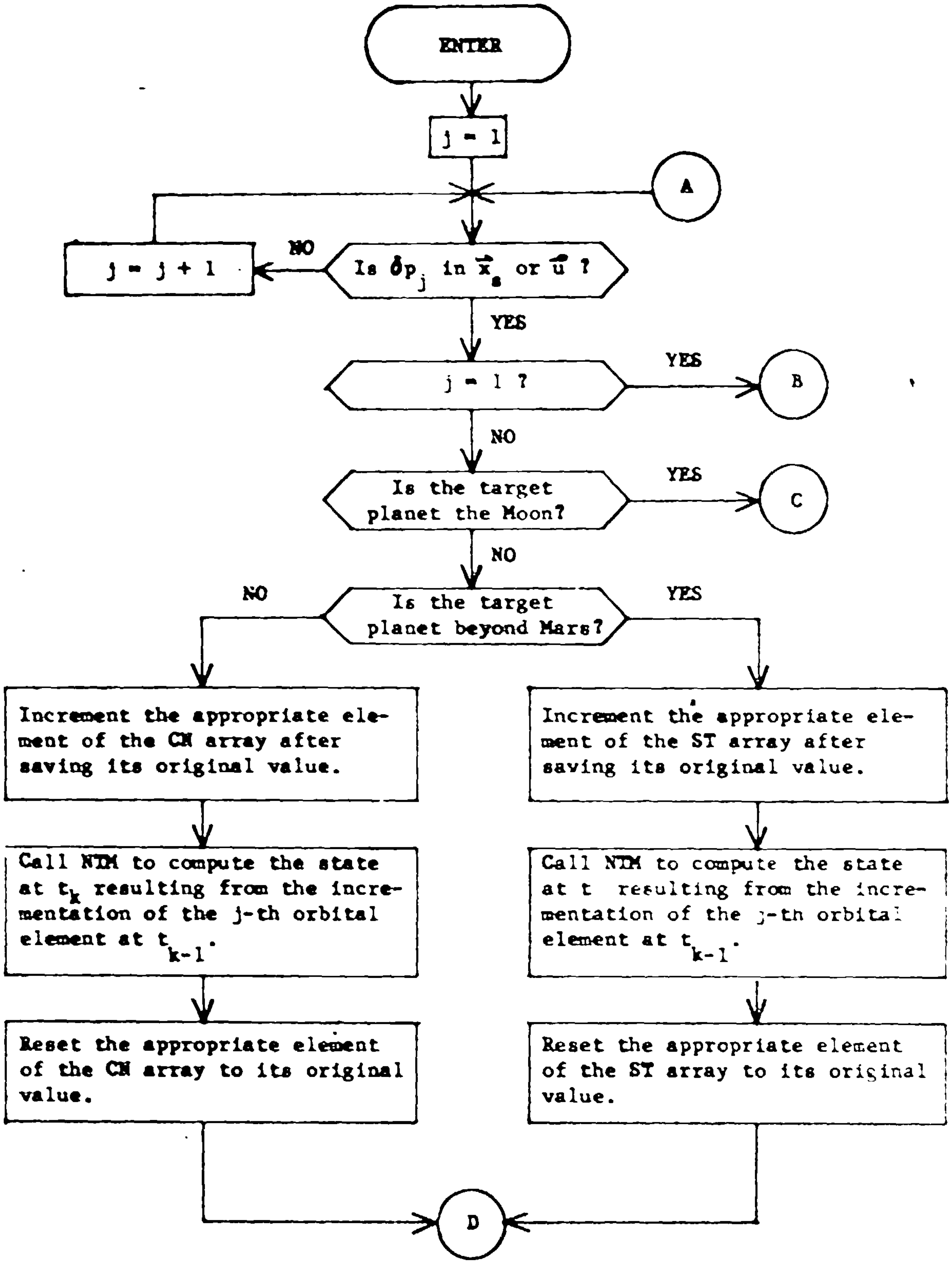
Thus, to increment Ω by $\Delta\Omega$ without changing the other 5 basic elements requires that we also increment $\tilde{\omega}$ by $\Delta\Omega$ for the case of a planet, and

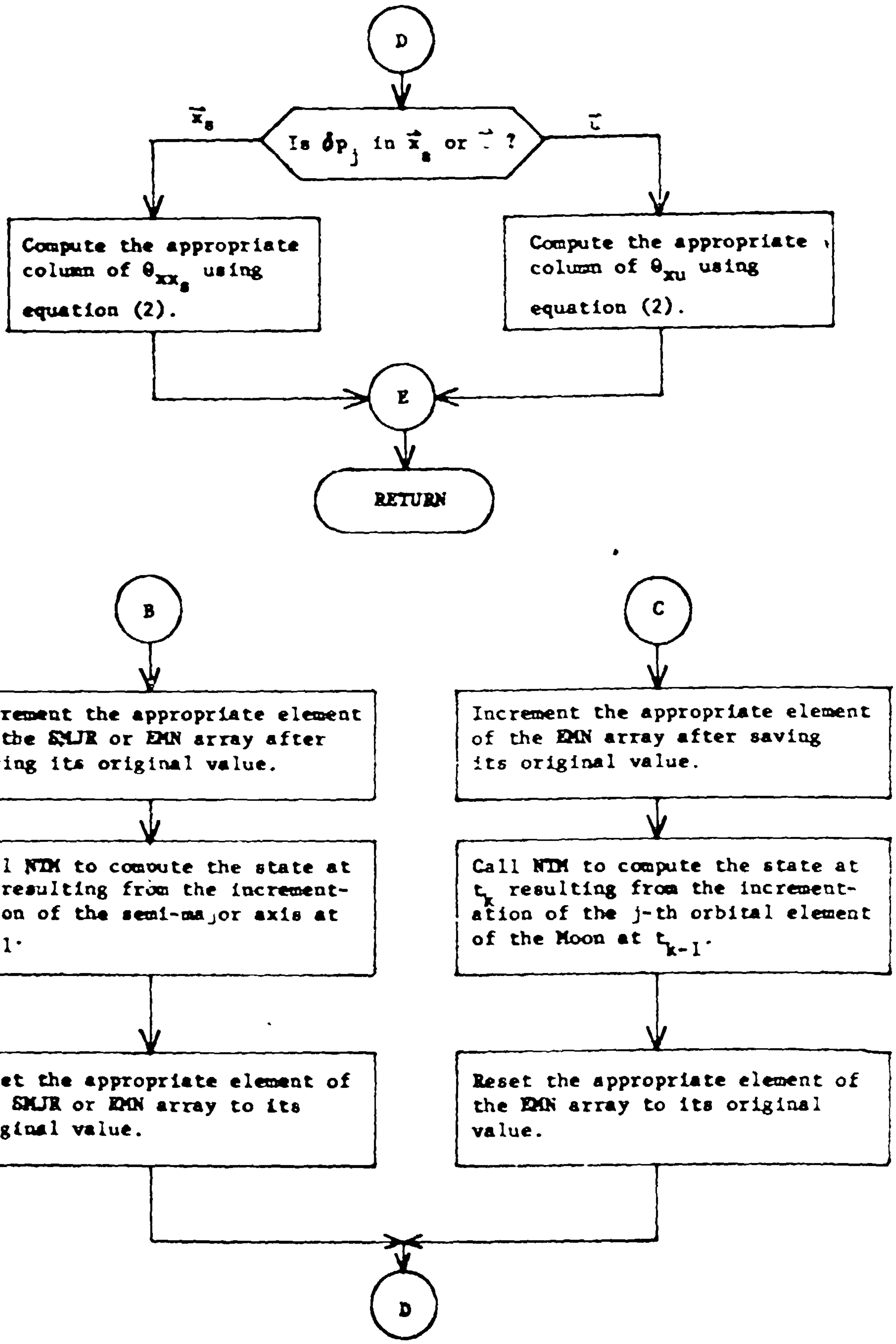
both $\tilde{\omega}$ and L by $\Delta\Omega$ for the case of the moon. To increment ω by $\Delta\omega$ we simply increment $\tilde{\omega}$ by $\Delta\omega$ for a planet, while for the moon we must increment both $\tilde{\omega}$ and L by $\Delta\omega$. To increment M by ΔM for the moon we simply increment L by ΔM.

In the PLND flow chart we employ the following definition:

$$P_j = \begin{cases} a & j = 1 \\ e & 2 \\ i & 3 \\ \Omega & 4 \\ \omega & 5 \\ M & 6 \end{cases}$$

PLND Flow Chart




D
$\vec{x}_s$
Is δp_j in $\vec{x}_s$ or $\vec{u}$?
$\vec{u}$
Compute the appropriate column of θ_{xx_s} using equation (2).
Compute the appropriate column of θ_{xu} using equation (2).
E
RETURN
B
C
Increment the appropriate element of the SMJR or EMN array after saving its original value.
Increment the appropriate element of the EMN array after saving its original value.
Call NTM to compute the state at t_k resulting from the incrementation of the semi-major axis at t_{k-1}.
Call NTM to compute the state at t_k resulting from the incrementation of the j-th orbital element of the Moon at t_{k-1}.
Reset the appropriate element of the SMJR or EMN array to its original value.
Reset the appropriate element of the EMN array to its original value.
D

SUBROUTINE POICOM

PURPOSE COMPUTE PROBABILITY OF IMPACT

CALLING SEQUENCE: CALL POICOM(XXXX,DET)

ARGUMENT:	XXXX	I	AIMPOINT IN THE IMPACT PLANE VECTOR
	DET	I	DETERMINANT OF LAMBDA MATRIX

SUBROUTINES SUPPORTED: BIAIM

SUBROUTINES REQUIRED: MATIN

LOCAL SYMBOLS:	PMQM	P+ MQM TRANSPOSE
	SAVE	INTERMEDIATE VARIABLE
	SUM	INTERME ATE VARIABLE
	W	ADA* PM * ADA TRANSPOSE

COMMON COMPUTED/USED:	IEND	POI	PSTAR	XLAM	
COMMON USED:	ADA	A	CR	EXEC	IIGP
	ONE	PI	PP	TWO	XLAMI
	ZERO				

POICOM Analysis

Subroutine POICOM computes the target condition covariance W_j^+ after a guidance correction, the projection of W_j^+ into the impact plane, and the probability of impact of the spacecraft with the target planet.

The target condition covariance matrix W_j^+ is defined as

$$W_j^+ = \eta_j \left(P_{k_j}^- + M\tilde{Q}_j M^T\right) \eta_j^T$$

where η_j is the variation matrix for the appropriate guidance policy, $P_{k_j}^-$ is the knowledge covariance prior to the guidance correction, $\tilde{Q}_j$ is the execution error covariance, and M is defined as the following 6 x 3 matrix:

$$M = \begin{bmatrix} 0 & \vdots & I \end{bmatrix}^T$$

Before the probability of impact can be computed, it is necessary to compute the projection Λ_j of W_j^+ into the impact plane. The covariance Λ_j is computed as follows for each of the three available midcourse guidance policies.

a. Fixed-time-of-arrival:

$$\Lambda_j = A\, W_j^+ A^T$$

where transformation A is defined in the subroutine BIAIM analysis.

b. Two-variable B-plane:

$$\Lambda_j = W_j^+$$

c. Three-variable B-plane:

$$\Lambda_j = \begin{bmatrix} 1 & 0 & 0 \\ 0 & 1 & 0 \end{bmatrix} W_j^+ \begin{bmatrix} 1 & 0 \\ 0 & 1 \\ 0 & 0 \end{bmatrix}$$

Assuming the probability density function associated with Λ_j is Gaussian and nearly constant over the target planet capture area permits us to compute the probability of impact using the equation

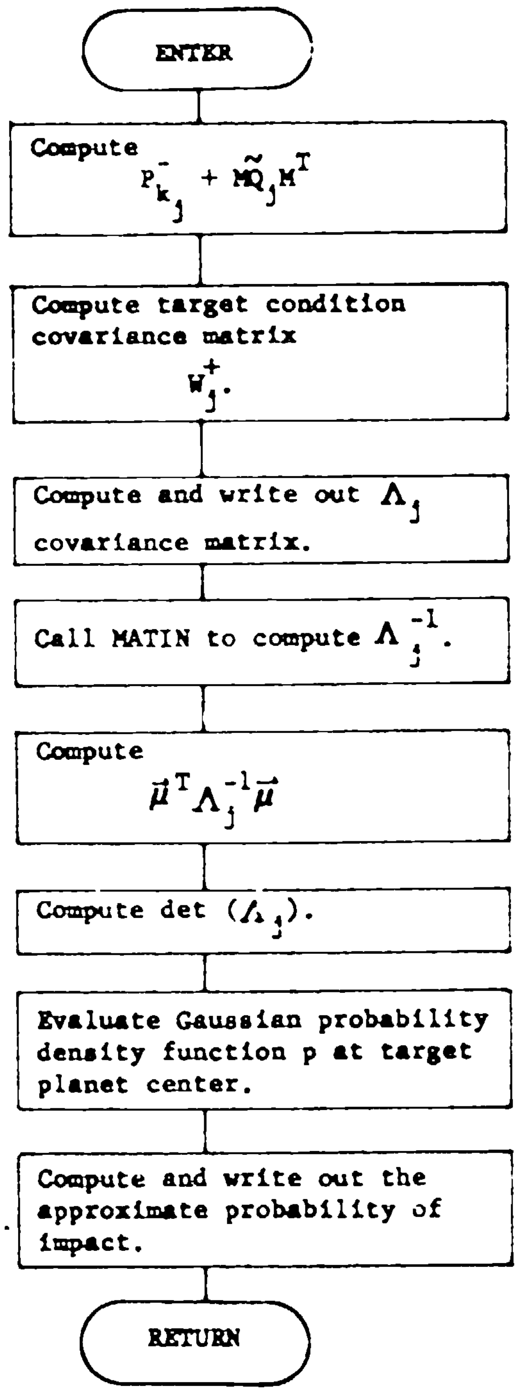
$$POI = \pi R_c^2 \, p$$

where R_c is the target planet capture radius and p is the Gaussian probability density function evaluated at the target planet center and given by

$$p = \frac{1}{2\pi |\Lambda_j|^{1/2}} \exp\left[-\frac{1}{2} \vec{\mu}^T \Lambda_j^{-1} \vec{\mu} \right]$$

where $\vec{\mu}$ is the aimpoint in the impact plane.

POICOM Flow Chart



PROGRAM PRED

PURPOSE CONTROL EXECUTION OF A PREDICTION EVENT IN THE ERROR ANALYSIS PROGRAM

SUBROUTINES SUPPORTED: ERRANN

SUBROUTINES REQUIRED: CORREL DYNO HYELS JACOBI NAVM NTM PSIM STMPR

LOCAL SYMBOLS

CXSU1	STORAGE FOR CXSU COVARIANCE ARRAY
CXSV1	STORAGE FOR CXSV COVARIANCE ARRAY
CXU1	STORAGE FOR CXU COVARIANCE ARRAY
CXV1	STORAGE FOR CXV COVARIANCE ARRAY
CXXS1	STORAGE FOR CXXS COVARIANCE ARRAY
DUM2	ARRAY OF EIGENVECTORS
DUM3	ARRAY OF EIGENVALUES
DUM	B DOT T AND B DOT R COVARIANCE MATRIX
EGVCT	ARRAY OF EIGENVECTORS
EGVL	ARRAY OF EIGENVALUES
ICODE	INTERNAL CONTROL FLAG
IPR	STORAGE FOR IPRINT
OUT	ARRAY OF STANDARD DEVIATIONS AND CORRELATION COEFFICIENTS
PEIG	MATRIX WHOSE HYPERELLIPSOID IS TO BE COMPUTED
PS1	STORAGE FOR PS COVARIANCE ARRAY
P1	STORAGE FOR P COVARIANCE ARRAY
RF	NOMINAL SPACECRAFT STATE AT TIME TPT
TPT	TIME TO WHICH PREDICTION IS TO BE MADE
VEIG	INTERMEDIATE VECTOR

COMMON COMPUTED/USED: CXSU CXSV CXU CXV CXXS IPRINT NPE PS P

COMMON COMPUTED:	DELTM	TRTM1	XI		
COMMON USED:	EM	FOP	FOV	IEIG	IHYP1
	ISTMC	NDIM1	NDIM2	NDIM3	NGE
	NTMC	ONE	Q	IPT2	TSOI1
	U0	V0	XF		

PRED Analysis

Subroutine PRED executes a prediction event in the error analysis mode. Subroutine PRED differs from subroutine PRESIM in two respects. First, the propagated knowledge covariance partitions are based on the (most recent) targeted nominal, rather than on the most recent nominal as in PRESIM. And second, estimated position/velocity deviations are not propagated in PRED since estimates are processed only in the simulation mode, and not in the error analysis mode. See subroutine PRESIM for further analytical details. A flow chart for PRED is not presented here since it is but a subset of the PRESIM flow chart.

SUBROUTINE PRELIM

PURPOSE TO PERFORM THE PRELIMINARY WORK ASSOCIATED WITH THE NOMNAL PROGRAM INCLUDING THE READING OF THE INPUT DATA, INITIALIZATION OF CONSTANTS, AND THE COMPUTATION OF A ZERO ITERATE IF REQUIRED

CALLING SEQUENCE: CALL PRELIM

SUBROUTINES SUPPORTED: NOMNAL

SUBROUTINES REQUIRED: CPMMS TIME ZERIT

LOCAL SYMBOLS:

DF	JULIAN DATE CORRESPONDING TO KALF ARRAY
DI	JULIAN DATE CORRESPONDING TO KALI ARRAY
GS	ARRAY OF VALUES OF SECONDS CORRESPONDING TO KALG ARRAY
I	INDEX
J	INDEX
KALF	CALENDAR DATE OF FINAL TRAJECTORY TIME
KALG	ARRAY OF CALENDAR DATES OF GUIDANCE EVENTS
KALI	CALENDAR DATE OF INITIAL TRAJECTORY TIME
KALT	ARRAY OF CALENDAR DATES OF TARGET TIMES
KEY	LOCAL VARIABLE USED TO COMPLETE INFORMATION IN THE ARRAY
SF	SECONDS OF FINAL TRAJECTORY TIME
SI	SECONDS OF INITIAL TRAJECTORY TIME
TS	SECONDS OF TARGET TIMES CORRESPONDING TO KALT ARRAY

COMMON COMPUTED/USED:

AC	ALNGTH	DG	D1	FI
IBADS	IBARY	ICOORD	IFINT	IPRE
ISTART	IZERO	KGYD	KMXQ	KOAST
KTIM	KTYP	LTARG	LVLS	MAT
MAXB	MDL	NBOD	NB	NCPR
NOGYD	NOIT	NPAR	ONE	PERV
PHILS	PSI1	PSI2	RIN	RPRAT
RP	SIGMAL	SPHFAC	SSS	TAR
THEDOT	THELS	TIMG	TIM1	TIM2
TIN	TMPR	TM	ZDAT	

COMMON COMPUTED:	DINTG	EIGHT	FIVE	FOUR	HALF
	IEPHEM	IPRINT	KSICA	KUR	NBODYI
	NINETY	RAD	TEN	THREE	TMU
	TRTM	TWO	ZERO		
COMMON USED:	ACKT	DELV	DT	DVMAX	IBAST
	KTAR	LEVELS	MAXBAD	NITS	NLP
	NTP	PHI	PMASS	TIMS	TOL

PRELIM Analysis

PRELIM is responsible for the preliminary work required by NOMNAL including the initialization of variables, the reading of input, and the computation of zero iterate values for initial time, position, and velocity if necessary.

On the first call to PRELIM, PRELIM presets constants to be used on the entire series of runs. These constants include the double precision numbers and the launch profile parameters. On subsequent calls these variables are not reset.

PRELIM then presets constants for individual runs. These constants presently include most of the guidance event parameters. The user may easily change the two sets of constants for his particular needs.

PRELIM then accepts the input data. It reads data in the NAMELIST format.

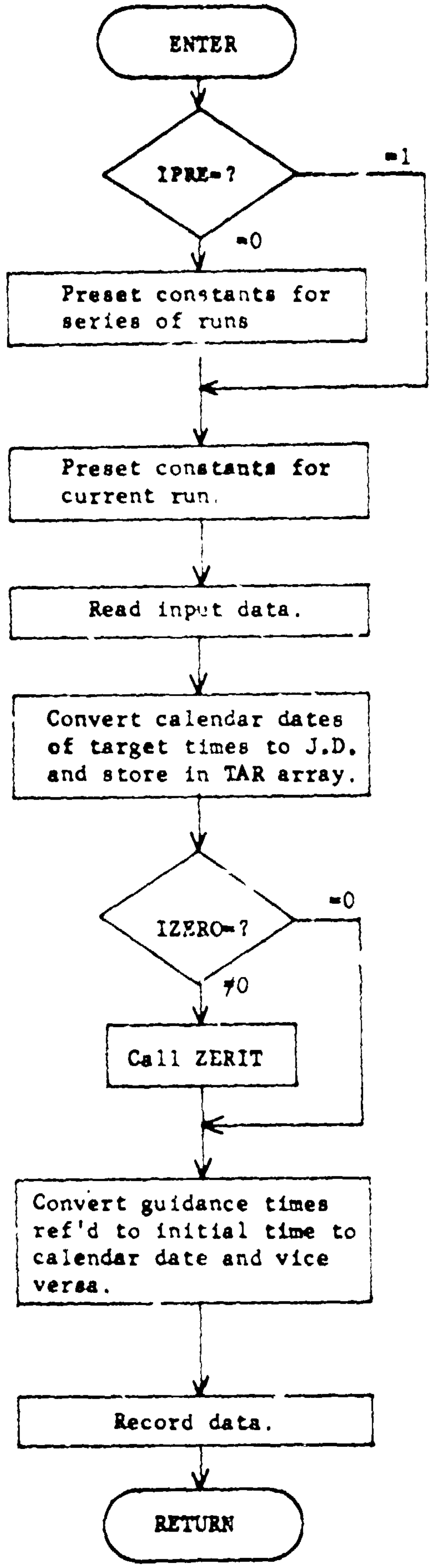
Target times must be read in as calendar dates. PRELIM next converts these to Julian date referenced 1900 and stores the converted values in the TAR array.

If the flag IZERO is nonzero, ZERIT is called for the computation of the zero iterate values of initial time, position, and velocity. ZERIT in turn calls HELIO for interplanetary trajectories and LUNA for lunar trajectories.

PRELIM then converts guidance event times referenced to initial time to calendar data and converts times read in as calendar dates to times referenced to the initial time. When the latter is done, it sets KTIM to acknowledge that conversion.

Finally PRELIM records all pertinent data.

PRELIM Flow Chart



SUBROUTINE PREPUL

PURPOSE: TO PERFORM THE PRELIMINARY COMPUTATIONS REQUIRED FOR THE PULSING ARC MODEL.

CALLING SEQUENCE: CALL PREPUL(RIN,DELTAV,D1)

ARGUMENTS:	RIN(6)	I	INERTIAL STATE OF SPACECRAFT AT NOMINAL TIME OF CORRECTION
	DELTAV(3)	I	TOTAL VELOCITY INCREMENT TO BE ADDED
	D1	I	JULIAN DATE OF NOMINAL TIME OF CORRECTION

SUBROUTINES SUPPORTED: EXCUTE EXCUTS

SUBROUTINES REQUIRED: TIME

LOCAL SYMBOLS:	A	SEMIMAJOR AXIS
	C	INTERMEDIATE VARIABLE IN F AND G SERIES
	DB	JULIAN DATE AT BEGINNING OF PULSING ARC
	DELVM	MAGNITUDE OF TOTAL IMPULSIVE CORRECTION
	DE	JULIAN DATE AT END OF PULSING ARC
	DVFM	MAGNITUDE OF FINAL PULSE OF SEQUENCE
	DVIM	MAGNITUDE OF TYPICAL PULSE OF SEQUENCE
	D	INTERMEDIATE VARIABLE IN F AND G SERIES
	G	GRAVITATIONAL CONSTANT OF BODY UNDER CONSIDERATION
	ID	CALENDAR DATE OF CRITICAL TIMES FOR OUTPUT
	MAXP	MAXIMUM NUMBER OF PULSES ALLOWED
	NDX	ARRAY OF CODES OF LAUNCH AND TARGET BODIES
	NX	INDEX OF GIVEN PLANET COORDINATES IN F-ARRAY
	RD	TIME DERIVATIVE OF RADIUS MAGNITUDE OF PLANET
	RR	MAGNITUDE OF RADIUS
	SD	SECONDS OF CRITICAL TIMES FOR OUTPUT

VV	SPEED OF PLANET				
COMMON COMPUTED/USED:	B	DVF	DVI	FS	GG
	GS	NPUL	PULT	RK	VK
COMMON USED:	ALNGTH	DTI	DUR	FIVE	FOUR
	F	NBOD	NB	NINETY	NLP
	NTP	PMASS	PULMAG	PULMAS	THREE
	TM	TWO	V		

PREPUL Analysis

PREPUL is responsible for performing the preliminary computations required for the pulsing arc model.

PREPUL first determines the nominal pulsing arc. Let the following definitions be made:

T	magnitude of pulsing engine thrust
m	nominal mass of spacecraft
Δt	duration of single pulse
Δt_i	time interval between pulses
$\overrightarrow{\Delta v}$	total velocity increment to be added

The velocity increment imparted by a single pulse is

$$\Delta v_i = \frac{T\,\Delta t}{m} \tag{1}$$

The number of pulses required is then

$$N_p = \left[\frac{\Delta v}{\Delta v_i}\right] + 1 \tag{2}$$

where $[\cdot]$ denotes the greatest integer function. The magnitude of the final pulse must be set to

$$\Delta v_f = \Delta v - (N_p - 1)\,\Delta v_i \tag{3}$$

The vector nominal pulse and final pulse are therefore given by

$$\begin{aligned} \overrightarrow{\Delta v}_i &= \Delta v_i \frac{\overrightarrow{\Delta v}}{\Delta v} \\ \overrightarrow{\Delta v}_f &= \Delta v_f \frac{\overrightarrow{\Delta v}}{\Delta v} \end{aligned} \tag{4}$$

The duration of the pulsing arc is then given by

$$\Delta T = (N_p - 1)\,\Delta t_i \tag{5}$$

Later computations require time histories of the position vectors of the launch and target bodies. An efficient means of obtaining this involves the f and g series. Given the state $\vec{r}_o$, $\vec{v}_o$ of body moving in a conic section about a central body of gravitational constant μ, the position vector as a function of t measured from the initial time is given by

$$\vec{r}(t) = f(t)\,\vec{r}_o + g(t)\,\vec{v}_o \tag{6}$$

where

$$f(t) = \sum_{k=0}^{n} f_k t^k \qquad g(t) = \sum_{k=1}^{n} g_k t^k \tag{7}$$

The constants f_k, g_k are computed in PREPUL as

$$f_o = 1$$

$$f_1 = 0$$

$$f_2 = \frac{-\mu}{2r_o^3}$$

$$f_3 = \frac{\mu \dot{r}_o}{2r_o^4}$$

$$f_4 = \frac{\mu^2}{24r_o^6} \left(4 - 15 \frac{r_o \dot{r}_o^2}{\mu} - 3 \frac{r_o}{a}\right)$$

$$f_5 = \frac{-\mu^2 \dot{r}_o}{8r_o^7} \left(4 - \frac{7r_o \dot{r}_o^2}{\mu} - 3 \frac{r_o}{a}\right)$$

$$f_6 = \frac{\mu^3}{720\, r_o^9} \left[- 70 + 114 \frac{r_o}{a} + 840 \frac{r_o \dot{r}_o^2}{\mu} - 630 \frac{r_o^2 \dot{r}_o^2}{\mu\, a} - 450 \left(\frac{r_o \dot{r}_o^2}{\mu}\right)^2 - 45 \frac{r_o^2}{a^2}\right]$$

$$g_1 = 1$$

$$g_2 = 0$$

$$g_3 = \frac{1}{3} f_2$$

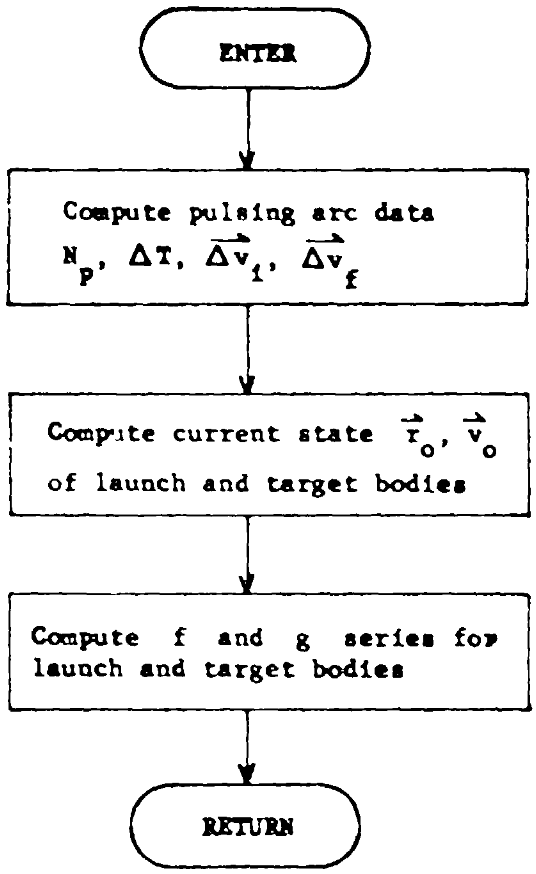
$$g_4 = \frac{1}{2} f_3$$

$$g_5 = \frac{3}{5} f_4 - \frac{1}{15} f_2^2$$

$$g_6 = \frac{2}{3} f_5 - \frac{1}{6} f_2 f_3$$

Reference: Baker, R. M. L. and Makemson, M. W., An Introduction to Astrodynamics, Academic Press, New York, 1967.

PREPUL Flow Chart



PROGRAM PRESIM

PURPOSE CONTROL EXECUTION OF A PREDICTION EVENT IN THE SIMULATION PROGRAM

SUBROUTINES SUPPORTED: SIMULL

SUBROUTINES REQUIRED: CORREL DYNOS HYELS JACOBI NAVM NTMS PSIM STMPR

LOCAL SYMBOLS

CXSU1	STORAGE FOR CXSU COVARIANCE ARRAY
CXSV1	STORAGE FOR CXSV COVARIANCE ARRAY
CXU1	STORAGE FOR CXU COVARIANCE ARRAY
CXV1	STORAGE FOR CXV COVARIANCE ARRAY
CXXS1	STORAGE FOR CXXS COVARIANCE ARRAY
DM2	ARRAY OF EIGENVECTORS
DM3	ARRAY OF EIGENVALUES
DM	B DOT T AND B DOT R COVARIANCE MATRIX
EGVCT	ARRAY OF EIGENVECTORS
EGVL	ARRAY OF EIGENVALUES
IPR	STORAGE FOR IPRINT
OUT	ARRAY OF STANDARD DEVIATIONS AND CORRELATION COEFFICIENTS
PEIG	MATRIX WHOSE HYPERELLIPSOID IS TO BE COMPUTED
PS1	STORAGE FOR PS COVARIANCE ARRAY
P1	STORAGE FOR P COVARIANCE ARRAY
RF1	MOST RECENT NOMINAL SPACECRAFT STATE AT TIME TP12
TPT	TIME TO WHICH PREDICTION IS TO BE MADE
VEIG	MATRIX TO BE DIAGONALIZED

COMMON COMPUTED/USED: CXSU CXSV CXU CXV CXXS ICODE IPRINT NPE PS P RI1

COMMON COMPUTED:	DELTM	RI	TRTM1	XI1	XI
COMMON USED:	ADEVXS	ADEVX	EDEVXS	EDEVX	EM
	FOP	FOV	IEIG	IHYF1	ISTMC
	NDIM1	NDIM2	NDIM3	NGE	NTMC
	ONE	PHI	Q	TEVN	TPT2
	TSOI1	TXXS	U0	V0	W
	XF1	XF	XSL	ZERO	

PRESIM Analysis

Subroutine PRESIM executes a prediction event in the simulation mode. In a prediction event the knowledge covariance partitions and the estimated position/velocity deviations from the most recent nominal trajectory are propagated forward to t_p, the time to which the prediction is to be made. The knowledge covariance partitions are propagated using the prediction equations found in the NAVM Analysis section. The estimate is propagated using the equation

$$\delta\tilde{\hat{X}}_p = \Phi(t_p, t_j)\ \delta\tilde{\hat{X}}_j + \Theta_{xx_s}(t_p, t_j)\ \delta\tilde{\hat{X}}_{s_j}$$

where Φ and Θ_{xx_s} are the state transition matrix partitions over the time interval $\left[t_j, t_p\right]$.

The position and velocity partitions of the propagated knowledge covariance are diagonalized at time t_p and the eigenvalues, eigenvectors, and hyper-ellipsoids are computed. If a guidance event has occurred previously, so that the matrix M relating B·T and B·R variations to position and velocity deviations at sphere of influence is available, and if t_p is within one day of the time t_{SI} at which the sphere of influence is encountered, then the B·T and B·R covariance matrix is also computed.

PRESIM Flow Chart

ENTER

Write out actual dynamic noise. Write out estimated and actual deviations from most recent nominal at prediction event time t_j .

Increment prediction event counter and obtain time t_p to which prediction is to be made.

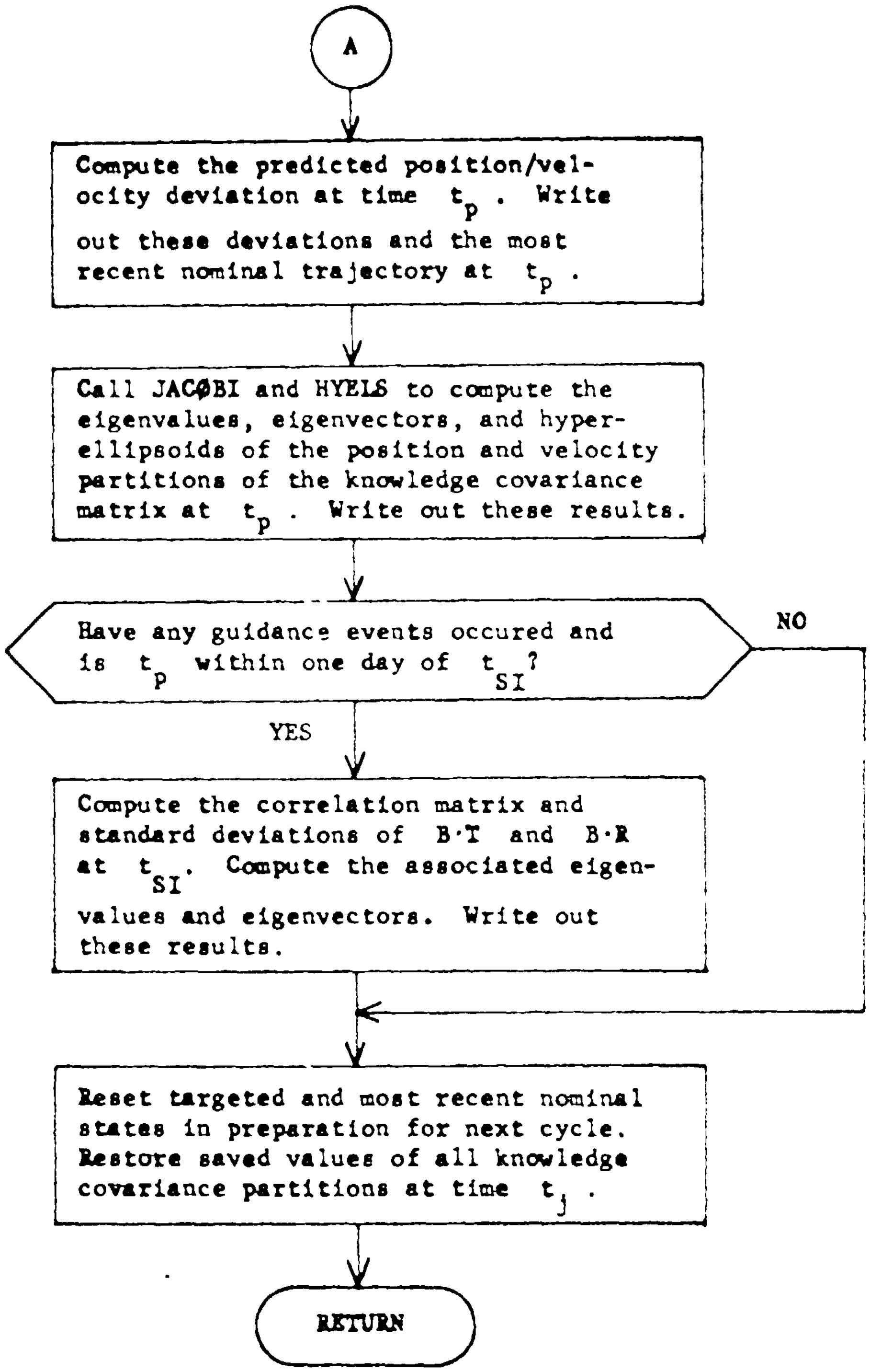
Save all knowledge covariance partitions at t_j .

Call NTMS to compute the most recent nominal trajectory at time t_p . Call PSIM to compute the state transition matrix partitions over the time interval $[t_j, t_p]$.

Call DYNØS to compute the dynamic noise covariance matrix for the interval $[t_j, t_p]$. Write out the state transition matrix partitions and the dynamic noise covariance matrix.

Call NAVM to propagate knowledge covariance partitions forward to time t_p . Write out the knowledge correlation matrix partitions and standard deviations at time t_p .

A


A
Compute the predicted position/velocity deviation at time t_p. Write out these deviations and the most recent nominal trajectory at t_p.
Call JACØBI and HYELS to compute the eigenvalues, eigenvectors, and hyper-ellipsoids of the position and velocity partitions of the knowledge covariance matrix at t_p. Write out these results.
Have any guidance events occured and is t_p within one day of t_{SI}?
NO
YES
Compute the correlation matrix and standard deviations of B·T and B·R at t_{SI}. Compute the associated eigenvalues and eigenvectors. Write out these results.
Reset targeted and most recent nominal states in preparation for next cycle. Restore saved values of all knowledge covariance partitions at time t_j.
RETURN

SUBROUTINE PRINT

PURPOSE: TO PRINT THE VIRTUAL MASS INFORMATION SPECIFIED.

CALLING SEQUENCE: CALL PRINT

SUBROUTINES SUPPORTED: VMP

SUBROUTINES REQUIRED: TIME TRAPAR NEWPGE SPACE

LOCAL SYMBOLS:

D	INTERMEDIATE VARIABLE USED FOR PRINTOUT
IDAY	DAY OF CALENDAR DATE OF CURRENT TIME
IHR	HOUR OF CALENDAR DATE OF CURRENT TIME
INCMNT	CURRENT TOTAL INCREMENTS FOR PRINTOUT
IP	CODE OF I-TH PLANET FOR PRINTOUT PURPOSES
IYR	YEAR OF CALENDAR DATE OF CURRENT TIME
MIN	MINUTES OF CALENDAR DATE OF CURRENT TIME
MO	MONTH OF CALENDAR DATE OF CURRENT TIME
RP	RADIUS OF I-TH PLANET RELATIVE TO INERTIAL FRAME
RS	RADIUS OF VEHICLE RELATIVE TO INERTIAL FRAME
RV	RADIUS OF VIRTUAL MASS RELATIVE TO INERTIAL FRAME
SEC	SECONDS OF CALENDAR DATE OF CURRENT TIME
TMP	POSITION AND VELOCITY OF VIRTUAL MASS RELATIVE TO PLANETS
VMR	MAGNITUDE OF VELOCITY OF VEHICLE RELATIVE TO VIRTUAL MASS
VP	MAGNITUDE OF VELOCITY OF I-TH PLANET FOR PRINTOUT PURPOSES
VS	MAGNITUDE OF VELOCITY OF VEHICLE RELATIVE TO INERTIAL FRAME
VSP	MAGNITUDE OF VELOCITY OF VEHICLE RELATIVE TO I-TH PLANET FOR PRINTOUT PURPOSES
VV	MAGNITUDE OF VELOCITY OF VIRTUAL MASS

RELATIVE TO INERTIAL FRAME

COMMON COMPUTED/USED:	F	V			
COMMON USED:	INCMNT	IPRT	NBODYI	NBODY	NO
	PLANET	ZERO			

SUBROUTINE PRINT3

PURPOSE: TO PRINT THE PERTINENT INFORMATION AT THE END OF EACH MEASUREMENT.

CALLING SEQUENCE: CALL PRINT3(MMCODE,NR)

ARGUMENT: MMCODE I MEASUREMENT CODE

NR I NUMBER OF ROWS IN THE OBSERVATION MATRIX H

SUBROUTINES SUPPORTED: ERRANN

SUBROUTINES REQUIRED: CORREL EPHEM ORB SYMPR TRAPAR

LOCAL SYMBOLS:

D	INTERMEDIATE DATE
IA	STATION NUMBER
D3	JULIAN DATE OF INITIAL TIME
D4	JULIAN DATE OF FINAL TIME
IDAY	CALENDAR DAY OF FINAL TIME
IHR	CALENDAR HOUR OF FINAL TIME
IMIN	CALENDAR MINUTE OF FINAL TIME
IMO	CALENDAR MONTH OF FINAL TIME
ITEMP	INTERMEDIATE VARIABLE
IYR	CALENDAR YEAR OF FINAL TIME
LDAY	CALENDAR DAY OF INITIAL TIME
LHR	CALENDAR HOUR OF INITIAL TIME
LMIN	CALENDAR MINUTES OF INITIAL TIME
LMO	CALENDAR MONTH OF INITIAL TIME
LYR	CALENDAR YEAR OF INITIAL TIME
M	NUMBER OF MEASUREMENT
RME	GEOCENTRIC RADIUS OF VEHICLE
RMP	DISTANCE OF VEHICLE FROM TARGET PLANET
SECI	CALENDAR SECONDS OF FINAL TIME

SECL	CALENDAR SECONDS OF INITIAL TIME
TRTM2	TRAJECTORY TIME AT END OF INTERVAL
VME	MAGNITUDE OF VELOCITY OF VEHICLE RELATIVE TO TO EARTH
VMP	MAGNITUDE OF VELOCITY OF VEHICLE RELATIVE TO TO TARGET PLANET

COMMON COMPUTED/USED:	NO	RE	RTP	XP	
COMMON USED:	AK	ALNGTH	AL	AM	CXSUP
	CXSU	CXSVP	CXSV	CXUP	CXU
	CXYP	CXV	CXXSP	CXXS	DATEJ
	DELTM	F	G	H	IBARY
	IPROB	IPRT	MCNTR	NMOD	NB
	NDIM1	NDIM2	NDIM3	NTP	PP
	PSP	PS	P	Q	R
	S	TM	TRTM1	U0	V0
	XF	XI	XLAB	XSL	XU
	XV				

SUBROUTINE PRINT4

PURPOSE: THIS SUBROUTINE PRINTS RELEVANT DATA AT THE END OF EACH MEASUREMENT IN THE SIMULATION MODE

CALLING SEQUENCE: CALL PRINT4(MMCODE,NR)

ARGUMENT: MMCODE I MEASUREMENT CODE

NR I NUMBER OF ROWS IN THE OBSERVATION MATRIX

SUBROUTINES SUPPORTED: SIMULL

SUBROUTINES REQUIRED: CORREL EPHEM ORB STMPR SUB1 TRAPAR

LOCAL SYMBOLS:

ADON	ACTUAL STATE DEVIATION FROM TARGETED NOMINAL TRAJECTORY
AODI	ACTUAL ORBIT ESTIMATION ERROR
D	INTERMEDIATE DATE
EDON	ESTIMATED STATE DEVIATION FROM TARGETED NOMINAL TRAJECTORY
IA	STATION NUMBER
IB	STAR*PLANET ANGLE NUMBER
M	MEASURMENT NUMBER
ROW	ARRAY OF CORRELATION COEFFICIENTS
SQP	VECTOR OF STANDARD DEVIATIONS
TRTM2	TRAJECTORY TIME AT END OF INTERVAL
XE1	POSITION AND VELOCITY OF EARTH AT TRTM1
XE2	POSITION AND VELOCITY OF EARTH AT TRTM2
XP1	POSITION AND VELOCITY OF TARGET PLANET AT TRTM1
XP2	POSITION AND VELOCITY OF TARGET PLANET AT TRTM2

COMMON COMPUTED/USED: NO

COMMON USED: ADEVXS ADEVX AK ALNGTH AL AM ANOIS AR AY CXSUP

CXSU	CXSVP	CXSV	CXUP	CXU
CXVP	CXV	CXXSP	CXXS	DATEJ
DELTM	EDEVXS	EDEVX	EY	F
G	HPHR	H	IBARY	IPROB
IPRT	MCNTR	NBOD	NB	NDIM1
NDIM2	NDIM3	NTP	PP	PSP
PS	P	Q	RES	R
S	TM	TRTM1	U0	V0
W	XF1	XF	XI1	XI
XLAB	XP	XSL	XU	XV
ZF	ZI			

PROGRAM PRNTS3

PURPOSE: TO PRINT A SUMMARY OF THE ERROR ANALYSIS MODE

SUBROUTINES SUPPORTED: ERRON

SUBROUTINES REQUIRED: CORREL TIME

LOCAL SYMBOLS:

D8	HOLLERITH LABEL INITIAL
D9	HOLLERITH LABEL FINAL
D1	JULIAN DATE, EPOCH JAN.0,1900, OF INITIAL TIME
D2	JULIAN DATE, EPOCH JAN. 0,1900, OF FINAL TIME
D3	JULIAN DATE OF INITIAL TIME
D4	JULIAN DATE OF FINAL TIME
F	FUNCTION= SQUARE ROOT OF SUM OF 3 SQUARES
IDAY	CALENDAR DAY OF FINAL TIME
IHR	CALENDAR HOUR OF FINAL TIME
IMIN	CALENDAR MINUTES OF FINAL TIME
IMO	CALENDAR MONTH OF FINAL TIME
IYR	CALENDAR YEAR OF FINAL TIME
LDAY	CALENDAR DAY OF INITIAL TIME
LHR	CALENDAR HOUR OF INITIAL TIME
LMIN	CALENDAR MINUTES OF INITIAL TIME
LMO	CALENDAR MONTH OF INITIAL TIME
LYR	CALENDAR YEAR OF INITIAL TIME
RI	POSITION AND VELOCITY OF VEHICLE AT INITIAL TIME
RMF	HELIOCENTRIC RADIUS OF VEHICLE AT FINAL TIME
RMI	HELIOCENTRIC RADIUS OF VEHICLE AT INITIAL TIME

SECI	CALENDAR SECONDS OF FINAL TIME
SECL	CALENDAR SECONDS OF INITIAL TIME
TRTM2	TRAJECTORY TIME AT END OF TRAJECTORY
VE	POSITION AND VELOCITY OF VEHICLE RELATIVE TO EARTH AT FINAL TIME
VMF	MAGNITUDE OF VELOCITY OF VEHICLE AT FINAL TIME
VMI	MAGNITUDE OF VELOCITY OF VEHICLE AT INITIAL TIME
VT	POSITION AND VELOCITY OF VEHICLE RELATIVE TO TARGET PLANET AT FINAL TIME

COMMON COMPUTED/USED: DC DSI

COMMON COMPUTED: TRTM1

COMMON USED:

ACCND	ACC	ALNGTH	BDRSI1	BDTSI1
BSI1	B	CXSUB	CXSU	CXSVB
CXSV	CXUB	CXU	CXVB	CXV
CXXSB	CXXS	DATEJ	DELTH	DELX
DNCN	DTMAX	FACP	FACV	FATH
IAUGIN	IDNF	IEPHEM	IMNF	IPROB
ISPH	ISTMC	ISTM1	MNCN	MNNAME
NDACC	NDIM1	NDIM2	NDIM3	NEV1
NEV2	NEV3	NEV	NMN	NST
NTMC	NTP	PB	PLANET	PSB
PS	P	RCA1	RE	RSOI1
RTP	SAL	SIGALP	SIGBET	SIGPRO
SIGRES	SLAT	SLON	TCA1	TM
TRTMB	TSOI1	UST	U0	VSOI1
VST	V0	WST	XB	XNM

PROGRAM PRNTS4

PURPOSE: TO PRINT OUT A SUMMARY OF THE SIMULATION MODE

SUBROUTINES SUPPORTED: SIMULL

SUBROUTINES REQUIRED: CORREL EPHEM ORB TIME NOMINAL

LOCAL SYMBOLS:

ADON	ACTUAL STATE DEVIATION FROM TARGETED NOMINAL TRAJECTORY AT FINAL TIME
AODI	ACTUAL ORBIT ESTIMATION ERROR AT FINAL TIME
BLANK	BLANK HOLLERITH CHARACTER
D1	JULIAN DATE, EPOCH JAN.0,1900, OF INITIAL TIME
D2	JULIAN DATE, EPOCH JAN.0,1900, OF FINAL TIME
D3	JULIAN DATE OF INITIAL TIME
D4	JULIAN DATE OF FINAL TIME
EDON	ESTIMATED STATE DEVIATION FROM TARGETED NOMINAL TRAJECTORY
IDAY	CALENDAR DAY OF FINAL TIME
IHR	CALENDAR HOUR OF FINAL TIME
IMIN	CALENDAR MINUTES OF FINAL TIME
IMO	CALENDAR MONTH OF FINAL TIME
IYR	CALENDAR YEAR OF FINAL TIME
LDAY	CALENDAR DAY OF INITIAL TIME
LHR	CALENDAR HOURS OF INITIAL TIME
LMIN	CALENDAR MINUTES OF INITIAL TIME
LMO	CALENDAR MONTH OF INITIAL TIME
LYR	CALENDAR YEAR OF INITIAL TIME
RE1	POSITION AND VELOCITY OF VEHICLE RELATIVE TO TO EARTH ON TARGETED NOMINAL

RE2	POSITION AND VELOCITY OF VEHICLE RELATIVE TO TO EARTH ON MOST RECENT NOMINAL
RE3	POSITION AND VELOCITY OF VEHICLE RELATIVE TO TO EARTH ON ACTUAL TRAJECTORY
RME1	GEOCENTRIC RADIUS OF VEHICLE ON TARGETED NOMINAL AT FINAL TIME
RME2	GEOCENTRIC RADIUS OF VEHICLE ON MOST RECENT NOMINAL AT FINAL TIME
RME3	GEOCENTRIC RADIUS OF VEHICLE ON ACTUAL TRAJECTORY AT FINAL TIME
RME	GEOCENTRIC RADIUS OF VEHICLE AT INITIAL TIME
RMP1	DISTANCE OF VEHICLE FROM TARGET PLANET ON TARGETED NOMINAL AT FINAL TIME
RMP2	DISTANCE OF VEHICLE FROM TARGET PLANET ON MOST RECENT NOMINAL AT FINAL TIME
RMP3	DISTANCE OF VEHICLE FROM TARGET PLANET ON ACTUAL TRAJECTORY AT FINAL TIME
RMP	DISTANCE OF VEHICLE FROM TARGET PLANET AT INITIAL TIME
RMS1	HELIOCENTRIC RADIUS OF VEHICLE AT FINAL TIME ON TARGETED NOMINAL
RMS2	HELIOCENTRIC RADIUS OF VEHICLE AT FINAL TIME ON MOST RECENT NOMINAL
RMS3	HELIOCENTRIC RADIUS OF VEHICLE AT FINAL TIME ON ACTUAL TRAJECTORY
RMS	HELIOCENTRIC RADIUS OF VEHICLE AT INITIAL TIME
RP1	STATE OF VEHICLE RELATIVE TO TARGET PLANET AT FINAL TIME ON TARGETED NOMINAL
RP2	STATE OF VEHICLE RELATIVE TO TARGET PLANET AT FINAL TIME ON MOST RECENT NOMINAL
RP3	STATE OF VEHICLE RELATIVE TO TARGET PLANET AT FINAL TIME ON ACTUAL TRAJECTORY
SECI	CALENDAR SECONDS AT FINAL TIME

SECL	CALENDAR SECONDS AT INITIAL TIME
VME	MAGNITUDE OF VELOCITY OF VEHICLE RELATIVE TO EARTH AT INITIAL TIME
VME1	MAGNITUDE OF VELOCITY OF VEHICLE RELATIVE TO EARTH ON TARGETED NOMINAL AT FINAL TIME
VME2	MAGNITUDE OF VELOCITY OF VEHICLE RELATIVE TO EARTH ON MOST RECENT NOMINAL AT FINAL TIME
VME3	MAGNITUDE OF VELOCITY OF VEHICLE RELATIVE TO EARTH ON ACTUAL TRAJECTORY AT FINAL TIME
VMP	MAGNITUDE OF VELOCITY OF VEHICLE RELATIVE TO TARGET PLANET AT INITIAL TIME
VMP1	MAGNITUDE OF VELOCITY OF VEHICLE RELATIVE TO TARGET PLANET ON TARGETED NOMINAL AT FINAL TIME
VMP2	MAGNITUDE OF VELOCITY OF VEHICLE RELATIVE TO TARGET PLANET ON MOST RECENT NOMINAL AT FINAL TIME TIME
VMP3	MAGNITUDE OF VELOCITY OF VEHICLE RELATIVE TO TARGET PLANET ON ACTUAL TRAJECTORY AT FINAL TIME
VMS	MAGNITUDE OF VELOCITY OF VEHICLE AT INITIAL TIME
VMS1	MAGNITUDE OF VELOCITY OF VEHICLE AT FINAL TIME ON TARGETED NOMINAL
VMS2	MAGNITUDE OF VELOCITY OF VEHICLE AT FINAL TIME ON MOST RECENT NOMINAL TRAJECTORY
VMS3	MAGNITUDE OF VELOCITY OF VEHICLE AT FINAL TIME ON ACTUAL TRAJECTORY

COMMON COMPUTED/USED:	NO	RE	RTP	XP	ZI
COMMON USED:	AALP	ABET	ACCND	ACC1	ACC
	ADEVXB	ADEVXS	ADEVX	ALNGTH	APRO
	ARES	AVARM	BDRSI1	BDRSI2	BDRSI3
	BDTSI1	BDTSI2	BDTSI3	BSI1	BSI2

BSI3	B	CXSUB	CXSU	CXSVB
CXSV	CXUB	CXU	CXVB	CXV
CXXSB	CXXS	DAB	DATEJ	DEB
DELMUP	DELMUS	DELX	DIB	DMAB
DMUPB	DMUSB	DNCN	DNOB	DTMAX
DWB	EDEVXS	EDEVX	FACP	FACV
FNTM	F	H	IAMNF	IAUGIN
IBARY	IDNF	IMNF	IPROB	ISOI1
ISOI2	ISOI3	ISTMC	ISTM1	MNCN
MNNAME	NBOD1	NBOD	NB1	NB
NDACC	NDIM1	NDIM2	NDIM3	NEV1
NEV2	NEV3	NEV5	NEV	NMN
NTMC	NTP	PB	PLANET	PSB
PS	P	RCA1	RCA2	RCA3
RSOI1	RSOI2	RSOI3	SAL	SIGALP
SIGBET	SIGPRO	SIGRES	SLAT	SLON
TCA1	TCA2	TCA3	TM	TRTMB
TSOI1	TSOI2	TSOI3	TTIM1	TTIM2
UNMAC	UST	UQ	VSOI1	VSOI2
VSOI3	VST	VQ	WST	XB
XF1	XF	XLAB	XNM	ZF

SUBROUTINE PSIM

PURPOSE: TO COMPUTE THE STATE TRANSITION MATRIX PARTITIONS PHI, TXXS, AND TXU OVER AN ARBITRARY INTERVAL OF TIME (TK, TK+1).

CALLING SEQUENCE: CALL PSIM(RI,RF,ISC)

ARGUMENT:	ISC	I	CODE SPECIFYING WHICH TECHNIQUE IS TO BE USED TO COMPUTE THE STATE TRANSITION MATRIX PARTITION PHI
	RF	I	POSITION AND VELOCITY OF THE VEHICLE AT THE END OF THE TIME INTERVAL
	RI	I	POSITION AND VELOCITY OF THE VEHICLE AT THE BEGINNING OF THE TIME INTERVAL

SUBROUTINES SUPPORTED: SIMULL SETEVS BIAIM GUISIM GUISS PRESIM ERRANN SETEVN GUIDM GUID PRED

SUBROUTINES REQUIRED: CASCAD CONC2 EPHEM MUND NDTM ORB PCTM PLND

LOCAL SYMBOLS:	D	INTERMEDIATE JULIAN DATE
	DELT	TIME INTERVAL IN CORRECT UNITS
	DUM	TEMPORARY STORAGE FOR STATE TRANSITION MATRIX
	IANS	VARIABLE USED IN EXAMINING IAUG ,IAUGIN
	POSS	DISTANCE OF THE VEHICLE FROM THE TARGET PLANET AT INITIAL TIME
	RS	POSITION OF VEHICLE RELATIVE TO GOVERNING BODY AT INITIAL TIME
	THSP	CONSTANT EQUAL TO SIX TIMES THE SPHERE OF INFLUENCE OF TARGET PLANET
	VEC	POSITION AND VELOCITY OF VEHICLE RELATIVE TO TARGET PLANET AT INITIAL TIME
	VS	VELOCITY OF VEHICLE RELATIVE TO GOVERNING BODY AT INITIAL TIME

COMMON COMPUTED/USED: NO XP

COMMON COMPUTED: PHI TXU TXXS

COMMON USED:	ALNGTH	DATEJ	DELTM	DTMAX	F
	IAUGDC	IAUG	IBARY	ISTM1	NBOD
	NB	NDIM1	NDIM2	NTP	RVS
	SPHERE	TM	TRTM1	VMU	ZERO

PSIM Analysis

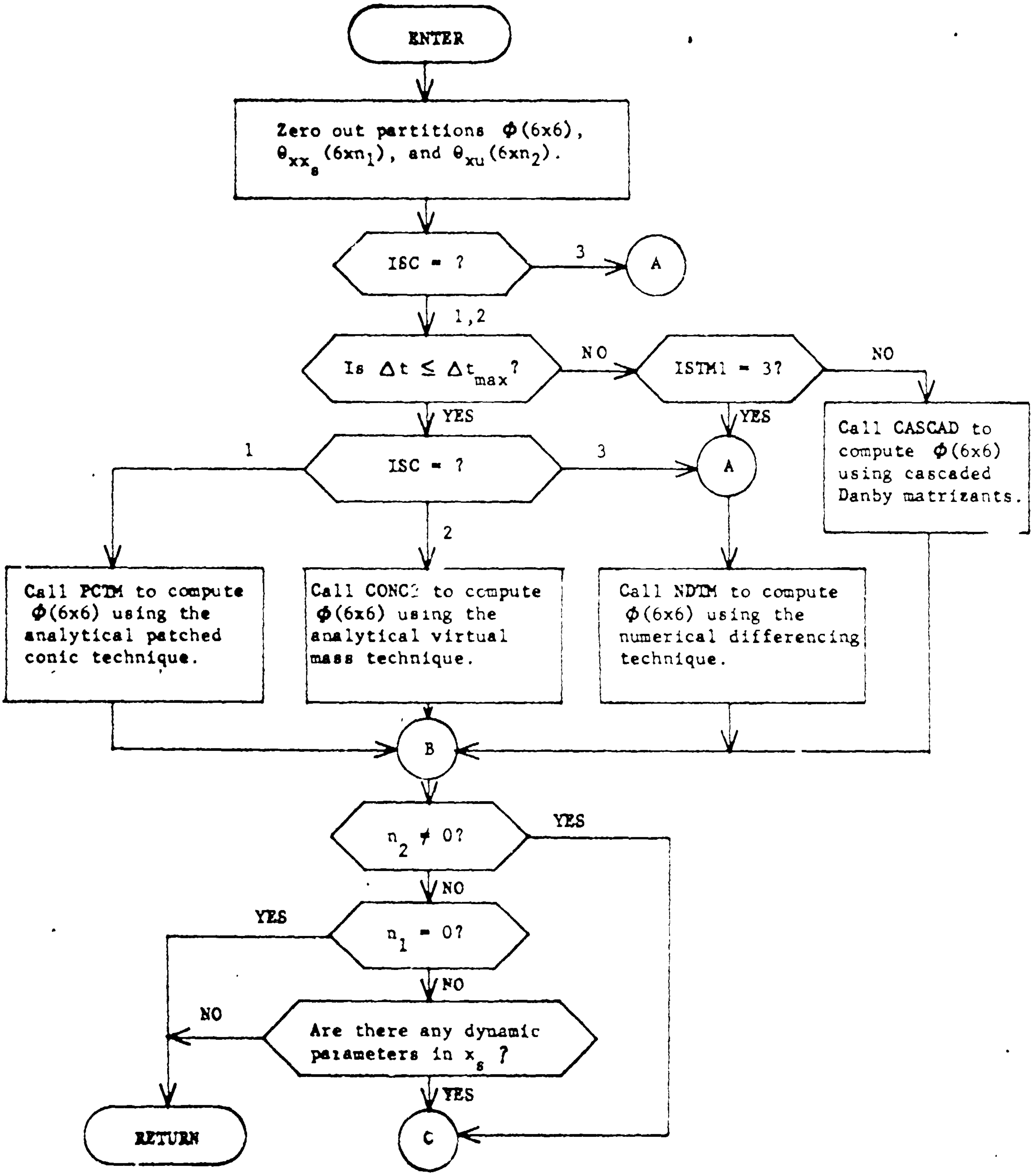
Subroutine PSIM controls the computation of each partition appearing in the augmented state transition matrix:

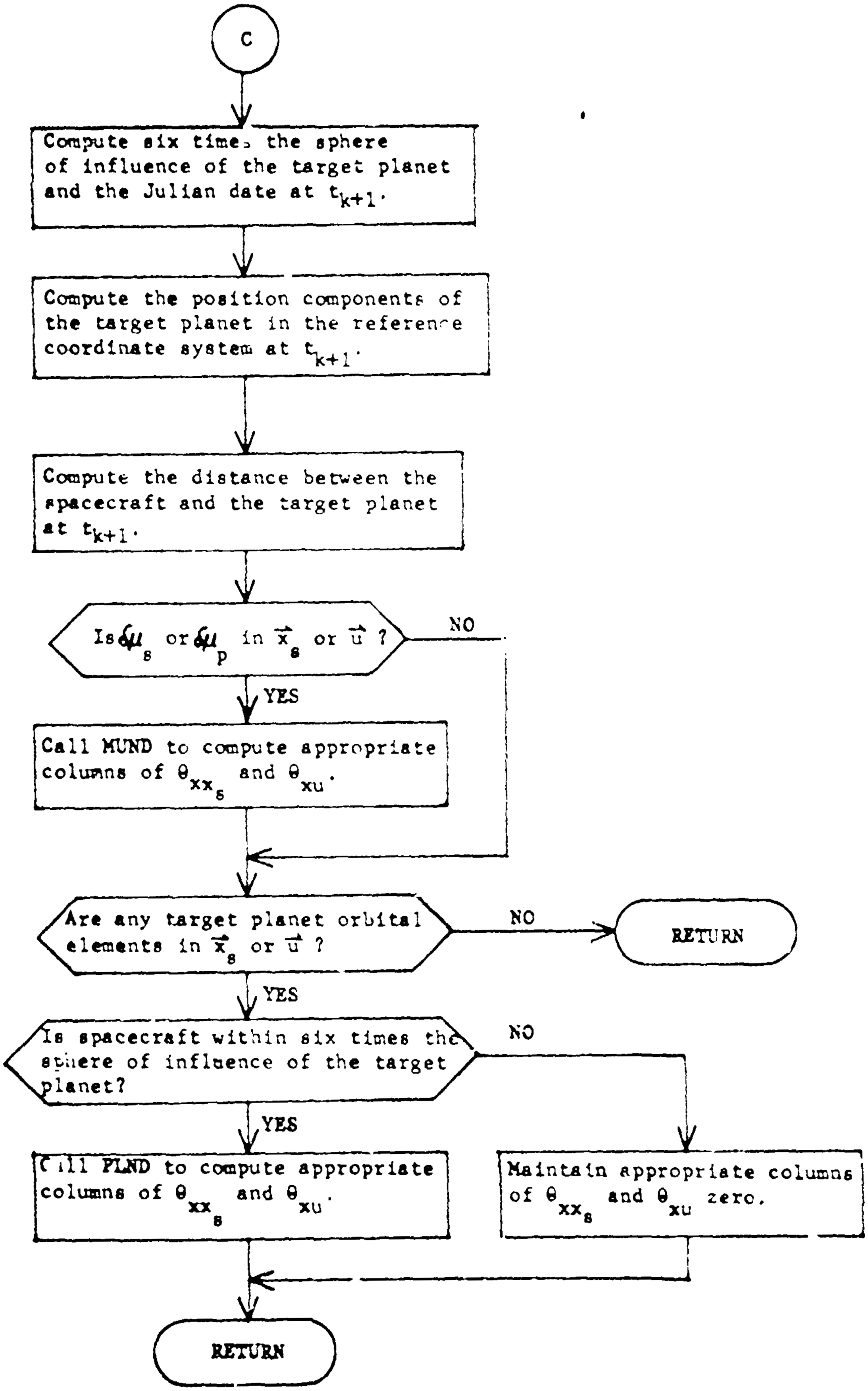
$$\Phi^A(k+1,k) = \begin{bmatrix} \Phi(k+1,k) & \theta_{xx_s}(k+1,k) & \theta_{xu}(k+1,k) & 0 \\ 0 & I & 0 & 0 \\ 0 & 0 & I & 0 \\ 0 & 0 & 0 & I \end{bmatrix}$$

The first part of the subroutine deals solely with the computation of $\Phi(k+1,k)$ by one of three techniques: analytical patched conic, analytical virtual mass, or numerical differencing. If an analytical technique is selected for computing $\Phi(k+1,k)$ over an interval of time greater than the maximum time interval for which the analytical technique is considered valid, we compute $\Phi(k+1,k)$ using numerical differencing or by cascading Danby matrizants.

The remaining partitions, θ_{xx_s} and θ_{xu}, are always computed by numerical differencing. Columns in these partitions associated with target planet gravitational constant or orbital elements are computed only if the spacecraft is within six times the sphere of influence of the target planet at t_{k+1}. Otherwise, these columns are set to zero.

PSIM Flow Chart




C
Compute six times the sphere of influence of the target planet and the Julian date at t_{k+1}.
Compute the position components of the target planet in the reference coordinate system at t_{k+1}.
Compute the distance between the spacecraft and the target planet at t_{k+1}.
Is μ_s or μ_p in $\vec{x}_s$ or $\vec{u}$?
NO
YES
Call MUND to compute appropriate columns of θ_{xx_s} and θ_{xu}.
Are any target planet orbital elements in $\vec{x}_s$ or $\vec{u}$?
NO
RETURN
YES
Is spacecraft within six times the sphere of influence of the target planet?
NO
YES
Call PLND to compute appropriate columns of θ_{xx_s} and θ_{xu}.
Maintain appropriate columns of θ_{xx_s} and θ_{xu} zero.
RETURN

SUBROUTINE PULCOV

PURPOSE COMPUTE EFFECTIVE EXECUTION ERROR COVARIANCE MATRIX FOR A VELOCITY CORRECTION MODELED AS AN IMPULSE SERIES

CALLING SEQUENCE: CALL PULCOV(RIN,DELTAV,TM,QK)

ARGUMENTS: RIN(6) I INERTIAL STATE OF SPACECRAFT AT NOMINAL TIME OF CORRECTION

DELTAV(3) I TOTAL VELOCITY INCREMENT TO BE ADDED

TM I TIME UNITS PER DAY

QK(6,6) O DEVIATION MATRIX RESULTING FROM EXECUTION ERRORS

SUBROUTINES SUPPORTED: EXCUT EXCUTS

SUBROUTINES REQUIRED: PERHEL QCOMP

LOCAL SYMBOLS: DELR PERTURBATION IN POSITION

DELV PERTURBATION IN VELOCITY

DVFM MAGNITUDE OF FINAL PULSE

DVIM MAGNITUDE OF TYPICAL PULSE

FSER F-SERIES CONSTANT FOR PLANET

GSER G-SERIES CONSTANT FOR PLANET

HLTF STATES OF LAUNCH AND TARGET BODIES AT END OF PROPAGATION INTERVAL

PERT CURRENT PERTURBATION

PHI STATE TRANSITION MATRIX OVER TYPICAL INTERVAL

QQ DEVIATION MATRIX DURING PROPAGATION THROUGH PULSES

Q TYPICAL VELOCITY EXECUTION ERROR COVARIANCE

RF NOMINAL INERTIAL STATE OF SPACECRAFT AT END OF TYPICAL INTERVAL

RPF PERTURBED INERTIAL STATE OF SPACECRAFT AT END OF TYPICAL INTERVAL

R INERTIAL STATE OF SPACECRAFT AT BEGINNING

	OF TYPICAL INTERVAL
T1	TIME INTERVAL BETWEEN PULSES
T2	T1**2
T3	T1**3
T4	T1**4
T5	T1**5
T6	T1**6

COMMON USED.

DTI	DVF	DVI	FS	GG
GS	NPUL	ONE	PSIGA	PSIGB
PSIGK	PSIGS	RK	TWO	VK
ZERO				

PULCØV Analysis

PULCØV processes the control covariance through the pulsing arc to determine a measure of the probabilistic deviation of the corrected trajectory from the desired trajectory resulting from execution errors.

The pulsing arc itself is computed in PREPUL. It consists of N_p-1 pulses $\overrightarrow{\Delta v}_i$ and a final pulse $\overrightarrow{\Delta v}_f$ satisfying

$$(N_p-1)\ \overrightarrow{\Delta v}_i + \overrightarrow{\Delta v}_f = \overrightarrow{\Delta v} \tag{1}$$

where $\overrightarrow{\Delta v}$ is the equivalent single impulse. The pulses are separated by a time interval Δt_i . The duration of the entire sequence of pulses is given by $\Delta T = (N_p-1)\ \Delta t_i$.

PULCØV must compute the execution error matrices Q, Q_f corresponding to the nominal pulse $\overrightarrow{\Delta v}_i$ and the final pulse $\overrightarrow{\Delta v}_f$ respectively. The error model for the engine is defined by the input specifications

σ_k^2 = proportionality error

σ_k^2 = resolution error

σ_α^2 = first pointing error

σ_β^2 = second pointing error

The execution error matrix measuring the probabilistic deviation of the actual velocity increment from the desired velocity increment is computed by QCØMP.

The exact equations defining the propagation of the covariance matrix are recursive in nature. If P_k^+ is the control covariance immediately after the k^{th} pulse, the covariance will propagate to the time of the next pulse t_{k+1} by the formula

$$P_{k+1}^- = \Phi_{k+1,k}\ P_k^+\ \Phi_{k+1,k}^T \tag{2}$$

where $\Phi_{k+1,k}$ is the 6x6 state transition matrix relating perturbations at t_{k+1} to perturbations at t_k . Adding the pulse at t_{k+1} expands the covariance by

$$P_{k+1}^{+} = P_{k+1}^{-} + \left[\begin{array}{c|c} 0 & 0 \\ \hline 0 & Q \end{array}\right] \tag{3}$$

where Q is set equal either to the nominal or final form of Q .

To start the process the control covariance following the first pulse is given by

$$P_{1}^{+} = \left[\begin{array}{c|c} 0 & 0 \\ \hline 0 & Q \end{array}\right] \tag{4}$$

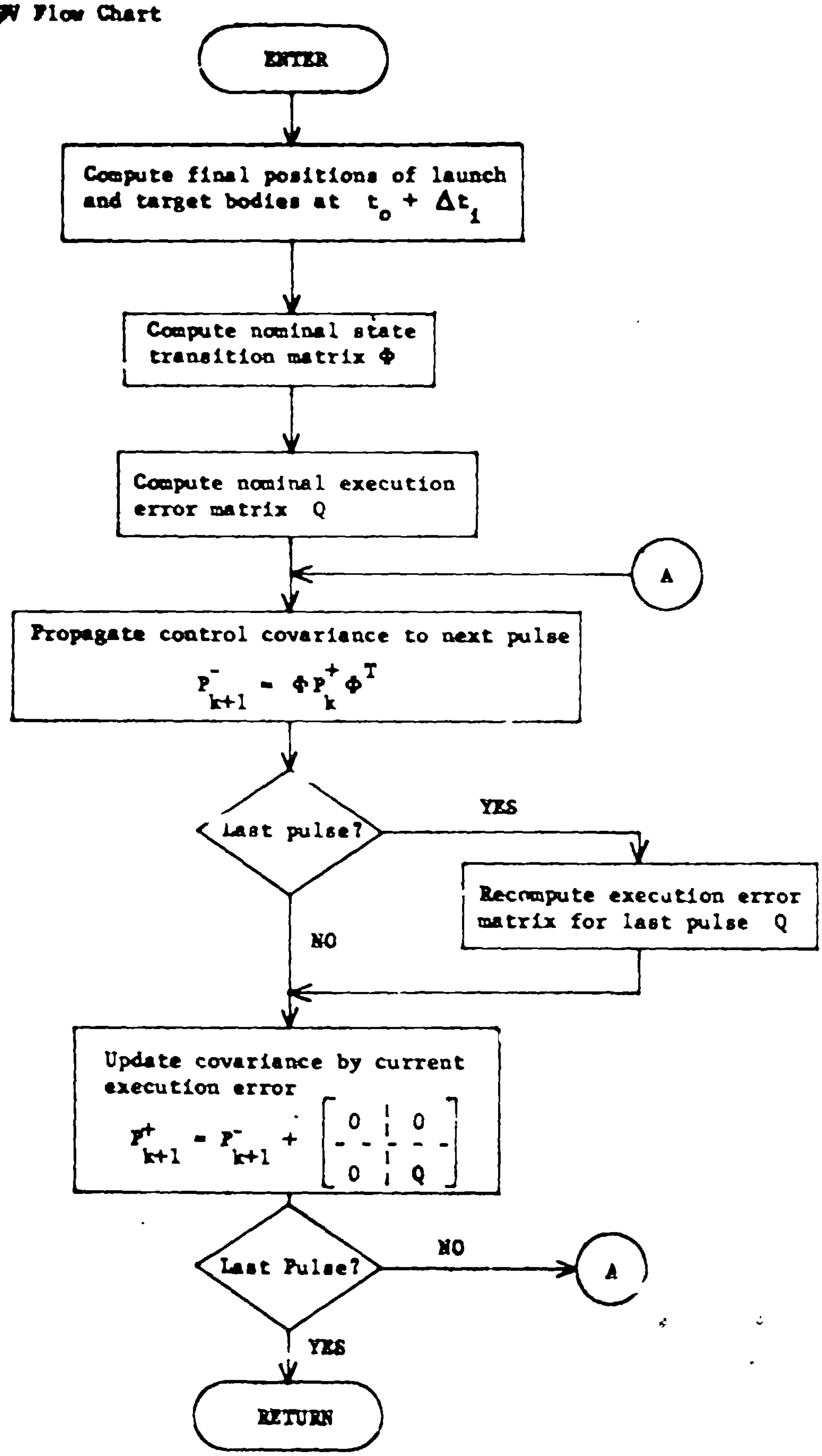
For efficiency one simplification is made in the process. Instead of recomputing the state transition matrix over each interval, the value of that matrix is held constant at the value corresponding to the "average interval". To explain this, let the state of the spacecraft at the time t_o of the impulsive $\overrightarrow{\Delta v}$ computation be denoted $\vec{r}_o$, $\vec{v}_o$. Then the "average interval" is defined to be the perturbed heliocentric trajectory (PERHEL) resulting from the propagation of the state $(\vec{r}_o, \vec{v}_o + \frac{1}{2}\overrightarrow{\Delta v})$ over the interval $(t_o, t_o + \Delta t_1)$.

The constant state transition matrix Φ is computed by numerical differencing. The initial state $(\vec{r}_o, \vec{v}_o + \frac{1}{2}\overrightarrow{\Delta v})$ is first propagated over the Δt_1 time interval (using PERHEL) resulting in the state $(\vec{r}_f, \vec{v}_f)$. Then the x-component of initial position is perturbed by Δx, leading to a final state of $(\vec{r}_{Pf}, \vec{v}_{Pf})$ upon propagation. The first column of the matrix is then computed by

$$\Phi_1 = \left[\frac{\vec{r}_{Pf} - \vec{r}_f}{\Delta x} , \frac{\vec{v}_{Pf} - \vec{v}_f}{\Delta x}\right]^T \tag{5}$$

The other columns of Φ are computed by similar computations using the remaining components of position and velocity $(y, z, \dot{x}, \dot{y}, \dot{z})$.

PULCOV Flow Chart



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SUBROUTINE   PULSEX

PURPOSE:  TO CONTROL EXECUTION OF THE PULSING ARC MODEL.

CALLING SEQUENCE:  CALL PULSEX(RIN,DELTAV,RE,TM,IRE)

ARGUMENTS:  RIN(6)      I INERTIAL STATE OF SPACECRAFT AT NOMINAL
                          TIME OF CORRECTION

            DELTAV(3)   I TOTAL VELOCITY INCREMENT TO BE ADDED

            RE(6)       O FINAL INERTIAL STATE OF SPACECRAFT (IRE)

            TM          I TIME UNITS PER DAY

            IRE         I FLAG DETERMINING FINAL STATE
                          =0  RETURN FINAL STATE AT END OF PULSE ARC
                          =1  RETURN FINAL STATE AT ARC MIDPOINT

SUBROUTINES SUPPORTED:  EXCUTE  EXCUTS  EXCUT

SUBROUTINES REQUIRED:   CAREL   PERHEL

LOCAL SYMBOLS:  A       SEMIMAJOR AXIS

                DTS     TIME INTERVAL IN TIME UNITS

                DT      DUMMY VARIABLE FOR OUTPUT

                E       ECCENTRICITY

                FSER    F-SERIES CONSTANT

                GSER    G-SERIES CONSTANT

                HLTF    STATES OF LAUNCH AND TARGET BODIES AT END
                        OF PROPAGATION INTERVAL

                HLTI    STATES OF LAUNCH AND TARGET BODIES AT
                        BEGINNING OF PROPAGATION INTERVAL

                IPUL    PULSE COUNTER

                PP      UNIT VECTOR TO PERIAPSIS

                QQ      UNIT VECTOR IN ORBITAL PLANE NORMAL TO PP

                RB      INERTIAL STATE OF SPACECRAFT AT BEGINNING
                        OF PROPAGATION INTERVAL

                TA      TRUE ANOMALY

                TFP     TIME FROM PERIAPSIS
```

TK	TIME FROM START OF PULSING ARC
TS	INTERMEDIATE VARIABLE
T1	TIME INTERVAL FROM MIDPOINT OF ARC
T2	T1**2
T3	T1**3
T4	T1**4
T5	T1**5
T6	T1**6
W	ARGUMENT OF PERIAPSIS
WW	UNIT NORMAL TO ORBITAL PLANE
XI	INCLINATION
XN	LONGITUDE OF ASCENDING NODE

COMMON USED:

DTI	DVF	DVI	F3	GG
GS	NPUL	ONE	PULT	RK
TWO	VK	ZERO		

PULSEX Analysis

PULSEX is responsible for the actual execution of the pulsing arc. Experiments have shown that adding an impulsive $\overrightarrow{\Delta v}$ at time t_o may be approximated quite closely by centering an equivalent sequence of smaller impulses about the nominal time t_o.

This equivalent sequence of thrusts is computed by PREPUL. It consists of $N_p - 1$ pulses $\overrightarrow{\Delta v_i}$ and a final pulse $\overrightarrow{\Delta v_f}$ satisfying

$$(N_p - 1)\ \overrightarrow{\Delta v_i} + \overrightarrow{\Delta v_f} = \overrightarrow{\Delta v} \tag{1}$$

The pulses are separated by a time interval Δt_i. The duration of the entire sequence of pulses is given by $\Delta T = (N_p - 1)\ \Delta t_i$.

For efficiency the perturbed heliocentric conic propagator PERHEL is used to propagate the trajectory between pulses. PERHEL requires the positions of the launch and target bodies at the beginning and end of each propagation interval. PREPUL stores the position and velocity of the launch and target bodies at the reference time t_o : $(\vec{r}_{LO}, \vec{v}_{LO})$ and $(\vec{r}_{TO}, \vec{v}_{TO})$ and stores the constants of the f and g series for those states $(f_{Lk}, g_{Lk}, f_{Tk}, g_{Tk}, k=1,6)$. The position of the launch body at some time t relative to the reference time t_o is then given by

$$\vec{r}_L(t) = f_L(t)\ \vec{r}_{LO} + g_L(t)\ \vec{v}_{LO} \tag{2}$$

where
$$f_L(t) = \sum_{k=0}^{6} f_{Lk}\ t^k$$
$$g_L(t) = \sum_{k=1}^{6} g_{Lk}\ t^k \tag{3}$$

with similar equations holding for the target body.

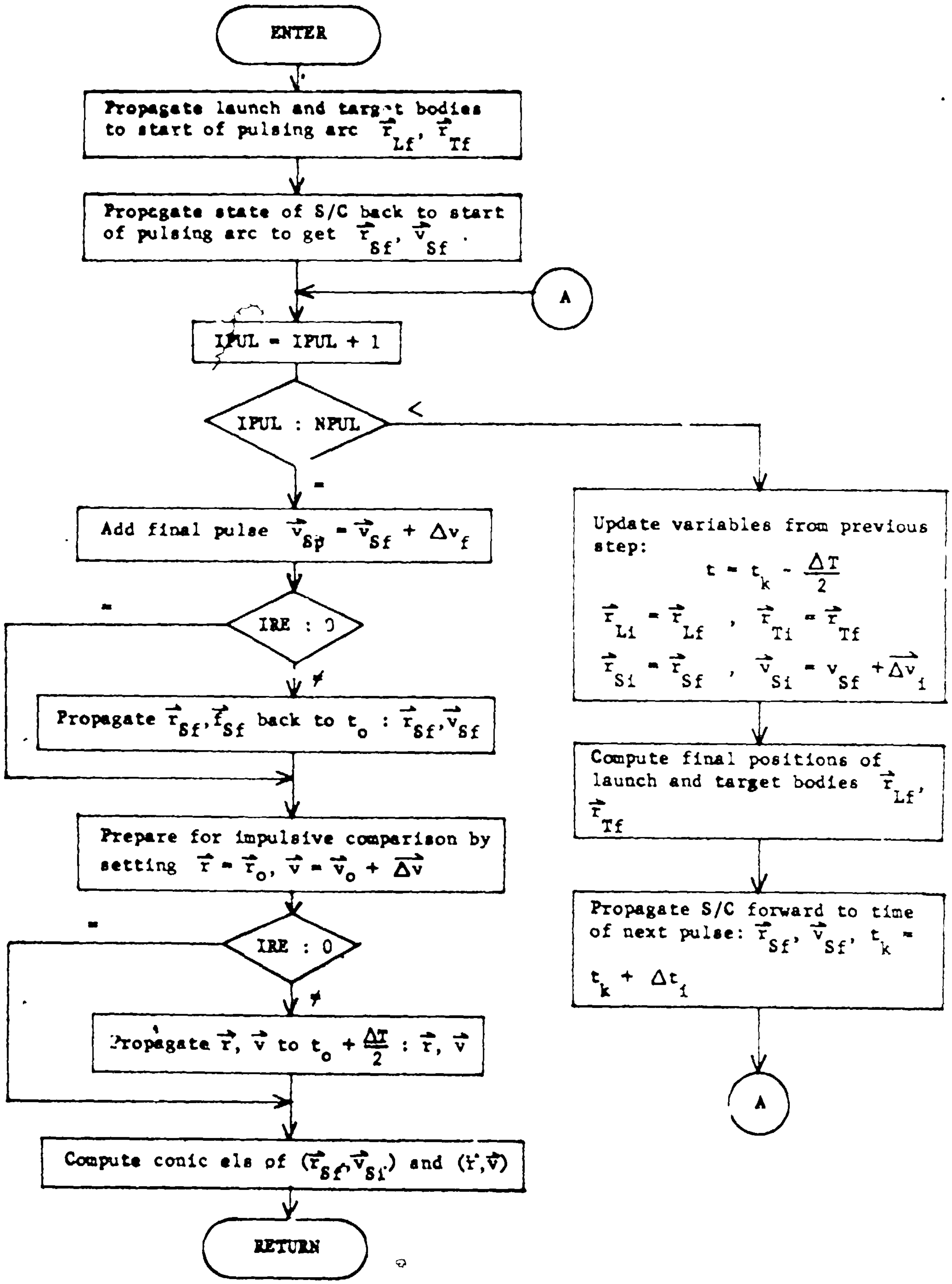
The procedure of PULSEX is straightforward. The positions of the launch and target bodies are computed at the time the pulsing arc should begin: $t_B = t_o - \Delta T/2$. PERHEL is then called to propagate the spacecraft from t_o backwards to t_B. The actual pulsing arc cycle is now entered. The nominal velocity increment $\overrightarrow{\Delta v_i}$ is added to the current velocity impulsively

$$\vec{v} = \vec{v} + \overrightarrow{\Delta v_i} \tag{4}$$

and the resulting state $(\vec{r}, \vec{v})$ is propagated forward over the time interval Δt_{ℓ} by PERHEL. Another pulse is added and the process repeated until $N_p - 1$ pulses have been added. Finally a pulse of $\overrightarrow{\Delta v}_f$ is added.

Two options are now permitted. If IRE = 0 , the final state is not altered (NOMINAL). If IRE = 1, the final state is propagated backwards back to t_o for use in ERRAN and SIMUL.

Finally CAREL is called to compute the conic elements of the final state. For comparison purposes, the impulsive $\overrightarrow{\Delta v}$ is added to the state at t_o, propagated to the final time $t_E = t_o + \Delta T/2$ by PERHEL, and those elements computed.


ENTER
Propagate launch and target bodies to start of pulsing arc $\vec{r}_{Lf}$, $\vec{r}_{Tf}$
Propagate state of S/C back to start of pulsing arc to get $\vec{r}_{Sf}$, $\vec{v}_{Sf}$.
A
IPUL = IPUL + 1
IPUL : NPUL
<
=
Add final pulse $\vec{v}_{Sp} = \vec{v}_{Sf} + \Delta v_f$
IRE : 0
=
≠
Propagate $\vec{r}_{Sf}, \vec{r}_{Sf}$ back to t_o : $\vec{r}_{Sf}, \vec{v}_{Sf}$
Prepare for impulsive comparison by setting $\vec{r} = \vec{r}_o$, $\vec{v} = \vec{v}_o + \vec{\Delta v}$
IRE : 0
=
≠
Propagate $\vec{r}$, $\vec{v}$ to $t_o + \frac{\Delta T}{2}$: $\vec{r}$, $\vec{v}$
Compute conic els of $(\vec{r}_{Sf}, \vec{v}_{Si})$ and $(\vec{r}, \vec{v})$
RETURN
Update variables from previous step:
$t = t_k - \frac{\Delta T}{2}$
$\vec{r}_{Li} = \vec{r}_{Lf}$, $\vec{r}_{Ti} = \vec{r}_{Tf}$
$\vec{r}_{Si} = \vec{r}_{Sf}$, $\vec{v}_{Si} = v_{Sf} + \vec{\Delta v}_i$
Compute final positions of launch and target bodies $\vec{r}_{Lf}$, $\vec{r}_{Tf}$
Propagate S/C forward to time of next pulse: $\vec{r}_{Sf}$, $\vec{v}_{Sf}$, $t_k = t_k + \Delta t_i$
A

SUBROUTINE QCOMP

PURPOSE: TO COMPUTE THE EXECUTION ERROR COVARIANCE MATRIX FOR A VELOCITY CORRECTION.

CALLING SEQUENCE: CALL QCOMP(V,EM,Q)

ARGUMENT:			
	V	I	VELOCITY CORRECTION
	Q	O	EXECUTION ERROR MATRIX
	EM	I	ERROR MODEL (SIGRES,SIGPRO,SIGALP,SIGBET)

SUBROUTINES SUPPORTED: BIAIM GUISIM PULCOV GUIDM

LOCAL SYMBOLS:		
	AU	SIGALP/U2
	BRK	SIGPRO+ SIGRES/R2
	BU	SIGBET/U2
	R2	U2+Z2
	U2	X2+Y2
	X2	V(1) SQUARED
	Y2	V(2) SQUARED
	Z2	V(3) SQUARED

QCØMP Analysis

Subroutine QCØMP computes the execution error covariance matrix $\tilde{Q}_j$ for a velocity correction $\overrightarrow{\Delta V} = (\Delta V_x, \Delta V_y, \Delta V_z)$ occurring at time t_j. If the execution error is assumed to have form

$$\delta\overrightarrow{\Delta V} = k\,\overrightarrow{\Delta V} + s\,\frac{\overrightarrow{\Delta V}}{\Delta V} + \delta\overrightarrow{\Delta V}_{\text{pointing}}$$

where k is the proportionality error and s is the resolution error, then the elements of the $\tilde{Q}_j$ matrix are given by

$$\tilde{Q}_{11} = \Delta V_x^{\,2}\left[\sigma_k^{\,2} + \frac{\sigma_s^{\,2}}{\rho^2}\right] + \frac{\Delta V_y^{\,2}\rho^2\sigma_{\delta\alpha}^2}{\mu^2} + \frac{\Delta V_x^{\,2}\Delta V_z^{\,2}\sigma_{\delta\beta}^2}{\mu^2}$$

$$\tilde{Q}_{12} = \tilde{Q}_{21} = \Delta V_x\Delta V_y\left[\sigma_k^{\,2} + \frac{\sigma_s^{\,2}}{\rho^2} - \frac{\rho^2\sigma_{\delta\alpha}^2}{\mu^2} + \frac{\Delta V_z^{\,2}\sigma_{\delta\beta}^2}{\mu^2}\right]$$

$$\tilde{Q}_{13} = \tilde{Q}_{31} = \Delta V_x\Delta V_z\left[\sigma_k^{\,2} + \frac{\sigma_s^{\,2}}{\rho^2} - \sigma_{\delta\beta}^{\,2}\right]$$

$$\tilde{Q}_{22} = \Delta V_y^{\,2}\left[\sigma_k^{\,2} + \frac{\sigma_s^{\,2}}{\rho^2}\right] + \frac{\Delta V_x^{\,2}\rho^2\sigma_{\delta\alpha}^2}{\mu^2} + \frac{\Delta V_y^{\,2}\Delta V_z^{\,2}\sigma_{\delta\beta}^2}{\mu^2}$$

$$\tilde{Q}_{23} = \tilde{Q}_{32} = \Delta V_y\Delta V_z\left[\sigma_k^{\,2} + \frac{\sigma_s^{\,2}}{\rho^2} - \sigma_{\delta\beta}^{\,2}\right]$$

$$Q_{33} = \Delta V_z^{\,2}\left[\sigma_k^{\,2} + \frac{\sigma_s^{\,2}}{\rho^2}\right] + \mu^2\sigma_{\delta\beta}^{\,2}$$

where $\mu^2 = \Delta V_x^{\,2} + \Delta V_y^{\,2}$, $\rho^2 = \mu^2 + \Delta V_z^{\,2}$ and $\sigma_s^{\,2}$, $\sigma_k^{\,2}$, $\sigma_{\delta\alpha}^2$, and $\sigma_{\delta\alpha}^2$ are the variances associated with the resolution, proportionality, and two pointing errors, respectively.

PROGRAM QUASI

PURPOSE: PERFORM QUASI-LINEAR FILTERING EVENT IN THE SIMULATION PROGRAM

CALLING SEQUENCE: CALL QUASI

SUBROUTINES SUPPORTED: SIMULL

LOCAL SYMBOLS:

COMMON COMPUTED/USED:	ADEVX	EDEVX	NQE	XF1	
COMMON COMPUTED:	TRTM1	XI1	XI		
COMMON USED:	ADEVXS	EDEVXS	NDIM1	TEVN	W
	XF	XSL	ZERO		

QUASI Analysis

At a quasi-linear filtering event the most recent nominal trajectory is updated by using the most recent state deviation estimate. If $\tilde{X}_j^-$ is the most recent nominal position/velocity state immediately preceding the event at time t_j, and if $\delta\hat{\tilde{X}}_j^-$ is the position/velocity deviation estimate, then immediately following the quasi-linear filtering event, the most recent nominal position/velocity state is given by

$$\tilde{X}_j^+ = \tilde{X}_j^- + \delta\hat{\tilde{X}}_j^-$$

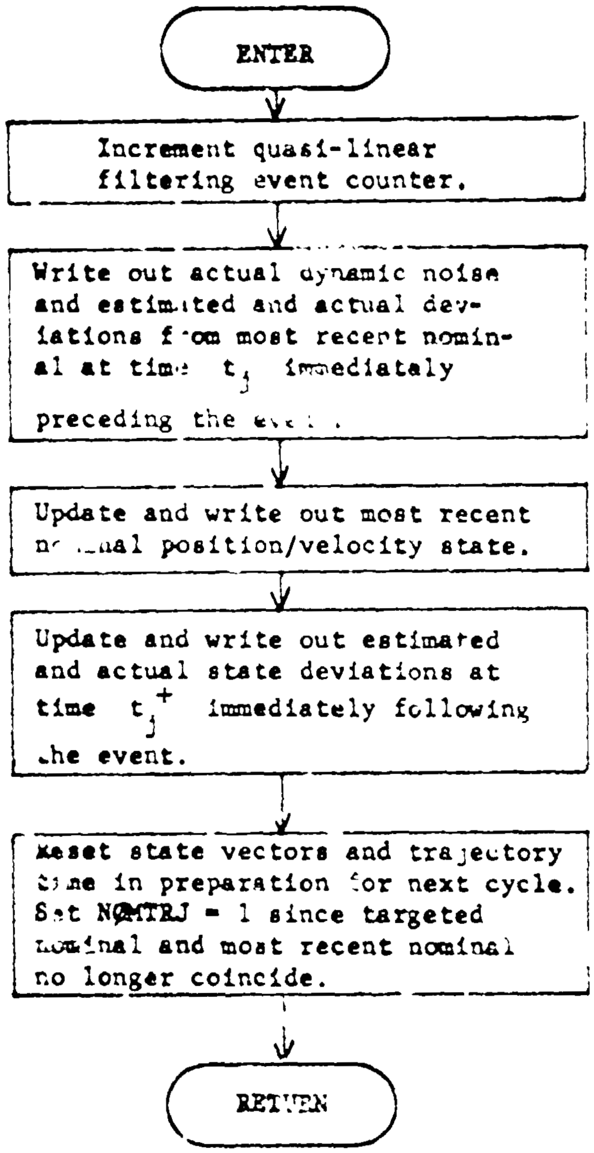
The estimated and actual deviations from the most recent nominal trajectory must also be updated:

$$\delta\hat{\tilde{X}}_j^+ = 0$$

$$\delta\tilde{X}_j^+ = \delta\tilde{X}_j^- - \delta\hat{\tilde{X}}_j^-$$

A quasi-linear filtering event in no way alters the knowledge and control uncertainties at time t_j. Thus knowledge covariance P_{k_j} and control covariance P_{c_j} remain constant across a quasi-linear filtering event. Furthermore, since no velocity correction is performed, the (most recent) targeted nominal $\overline{X}_j$ is unchanged. Neither is the solve-for parameter state updated at a quasi-linear filtering event.

QUASI Flow Chart



FUNCTION RNUM

PURPOSE: TO RETURN RANDOM NUMBERS ON A NORMAL DISTRIBUTION WITH MEAN ZERO AND STANDARD DEVIATION SIGMA.

CALLING SEQUENCE: Z=RNUM(SIGMA)

ARGUMENT: SIGMA I STANDARD DEVIATION

SUBROUTINES SUPPORTED: SIMULL

LOCAL SYMBOLS:

A	SUM OF TWELVE RANDOM NUMBERS BETWEEN ZERO AND ONE
NX	CONTROLLING INTEGER
N	INTERMEDIATE INTEGER
Q	INTERMEDIATE VARIABLE
RNUM	RANDOM NUMBER FROM NORMAL DISTRIBUTION WITH MEAN ZERO AND STANDARD DEVIATION SIGMA
RR	INTERMEDIATE VARIABLE
SS	INTERMEDIATE VARIABLE
WW	INTERMEDIATE VARIABLE
W1	INTERMEDIATE VARIABLE
YY	INTERMEDIATE VARIABLE
Y1	INTERMEDIATE VARIABLE
ZZ	INTERMEDIATE VARIABLE
Z1	INTERMEDIATE VARIABLE

RNUM Analysis

Function subprogram RNUM supplies random numbers on a normal distribution with mean zero and standard deviation σ .

Twelve random numbers X_i between 0 and 1 are computed, which are then used to compute the returned random number RNUM using the following equation:

$$RNUM = \left[\sum_{i=1}^{12} X_i - 6 \right] \cdot \sigma$$

SUBROUTINE SCHED

PURPOSE: TO DETERMINE WHAT TYPE OF MEASUREMENT IS TO BE TAKEN NEXT AND AT WHAT TIME IT WILL OCCUR.

CALLING SEQUENCE: CALL SCHED(T1,T2,MMCODE)

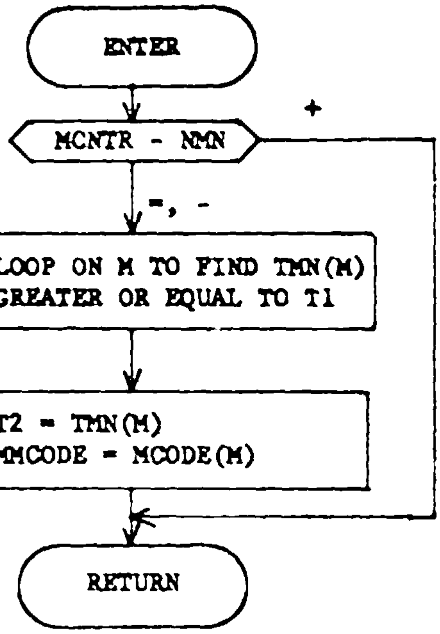
ARGUMENT:	MMCODE	O	MEASUREMENT MODEL CODE
	T1	I	PRESENT TRAJECTORY TIME
	T2	O	TRAJECTORY TIME AT WHICH THE NEXT MEASUREMENT OCCURS

SUBROUTINES SUPPORTED: SIMULL ERRANN

LOCAL SYMBOLS: M INDEX

COMMON USED: MCNTR MCODE NMM TMN

SCHED Flow Chart



SUBROUTINE SERIE

PURPOSE: TO COMPUTE THE TRANSCENDENTAL FUNCTIONS USED IN FLITE.

CALLING SEQUENCE: CALL SERIE(X,SX,CX)

ARGUMENTS:	X	I	INDEPENDENT VARIABLE
	SX	O	BATTIN S-FUNCTION OF X
	CX	O	BATTIN C-FUNCTION OF X

SUBROUTINES SUPPORTED: FLITE

SUBROUTINES REQUIRED: NONE

LOCAL SYMBOLS:	COSH	STATEMENT FUNCTION FOR HYPERBOLIC COSINE
	SINH	STATEMENT FUNCTION FOR HYPERBOLIC SINE
	E	SQRT OF ABS VALUE OF X

SERIE Analysis

SERIE computes the transcendental functions S(x) and C(x) used in the FLITE program in the solution of Lambert's theorem.

The functions S(x) and C(x) are defined by

$$S(x) = \frac{\sqrt{x} - \sin\sqrt{x}}{\sqrt{x}^{3}} \qquad x > 0$$

$$= \frac{\sinh\sqrt{-x} - \sqrt{-x}}{\sqrt{-x}^{3}} \qquad x < 0$$

$$= \frac{1}{6} \qquad x = 0 \qquad (1)$$

$$C(x) = \frac{1 - \cos\sqrt{x}}{x} \qquad x > 0$$

$$= \frac{\cosh\sqrt{-x} - 1}{-x} \qquad x < 0$$

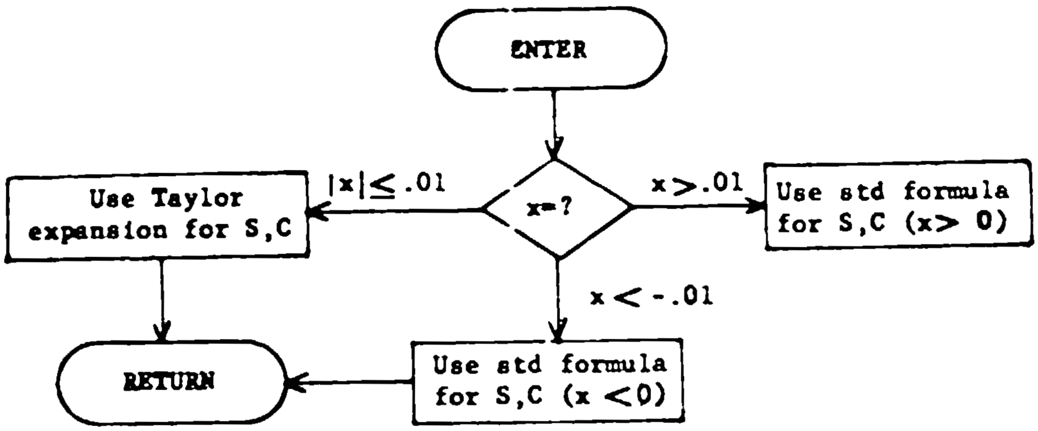
$$= \frac{1}{2} \qquad x \quad 0 \qquad (2)$$

For small values of $|x|$ the Taylor series expansions are used

$$S(x) = \frac{1}{3!} - \frac{x}{4!} + \frac{x^2}{5!} + \ldots$$

$$C(x) = \frac{1}{2!} - \frac{x}{3!} + \frac{x^2}{4!} + \ldots \qquad (3)$$

SERIE Flow Chart



SUBROUTINE SETEVN

PURPOSE PERFORM ALL COMPUTATIONS COMMON TO MOST EVENTS IN THE ERROR ANALYSIS PROGRAM

CALLING SEQUENCE: CALL SETEVN(RI,TEVN,NCODE)

ARGUMENT:	NCODE	I	EVENT CODE
	RI	I	TARGETED NOMINAL SPACECRAFT STATE AT PREVIOUS MEASUREMENT OR EVENT TIME
	TEVN	I	EVENT TIME

SUBROUTINES SUPPORTED: ERRANN

SUBROUTINES REQUIRED: CORREL DYNO HYELS JACOBI NAVM NTM PSIM STMPR TITLE TRAPAR

LOCAL SYMBOLS	BLANK	DUMMY CALLING ARGUMENT
	EGVCT	ARRAY OF EIGENVECTORS CORRESPONDING TO THE COLUMNS OF A GIVEN MATRIX
	EGVL	ARRAY OF EIGENVALUES RELATED TO THE EIGENVECTORS CONTAINED IN EGVCT
	ICODE	INTERNAL CONTROL FLAG
	OUT	SQUARE ROOTS OF EIGENVALUES
	PEIG	MATRIX FOR WHICH HYPERELLIPSOID IS TO BE COMPUTED
	RF	NOMINAL SPACECRAFT STATE AT EVENT TIME
	VEIG	MATRIX TO BE DIAGONALIZED

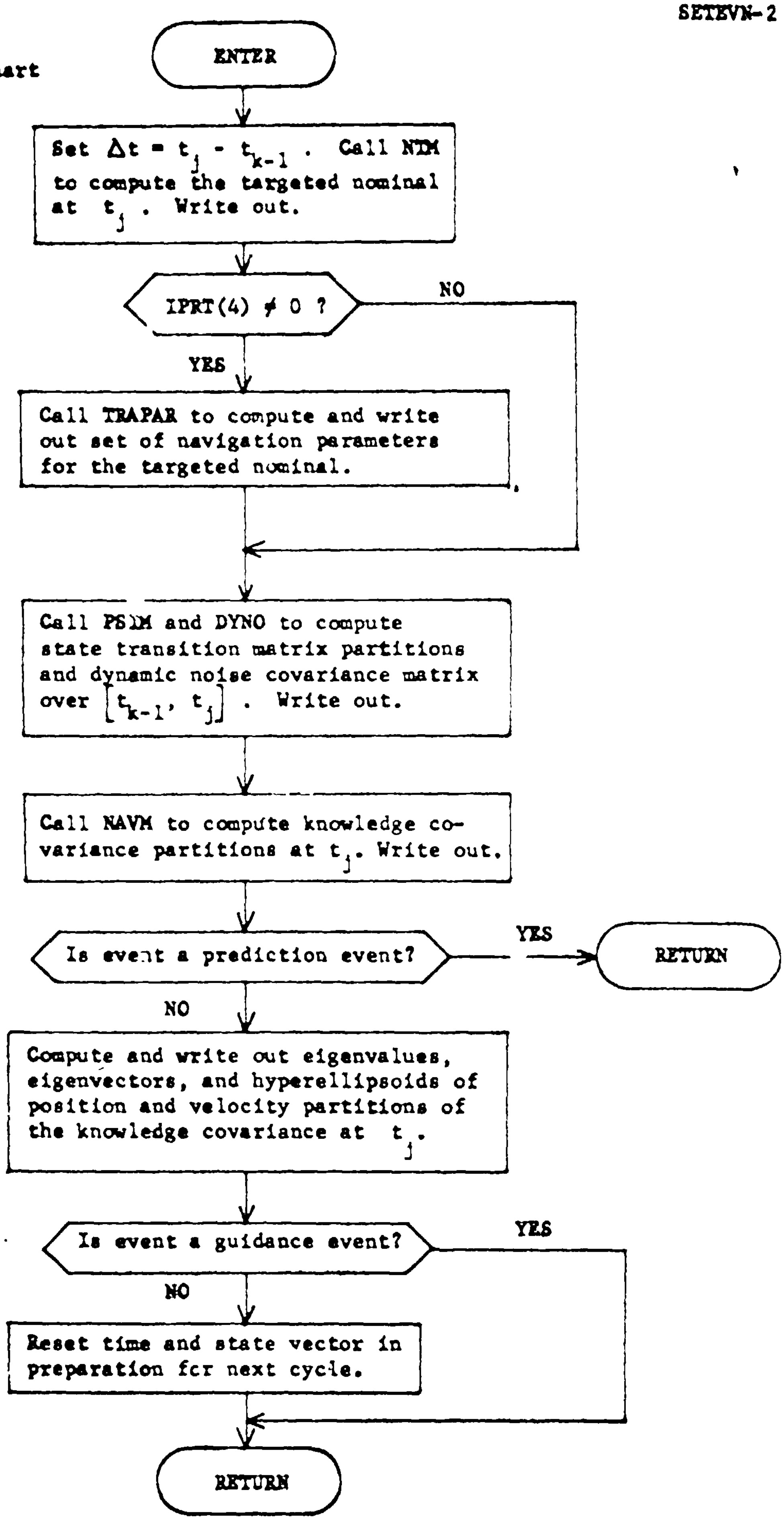
COMMON COMPUTED/USED:	TRTM1	XF			
COMMON COMPUTED:	DELTM	XI			
COMMON USED:	CXSU	CXSV	CXU	CXV	CXXS
	FOP	FOV	IEIG	IHYP1	IPRT
	ISTMC	NTMC	PS	P	Q
	U0	V0	XLAB		

SETEVN Analysis

Prior to executing any event in the error analysis mode, subroutine SETEVN is called to perform a series of computations which are common to all events. Subroutine SETEVN computes the targeted nominal trajectory at t_j, and propagates the knowledge covariance partitions at t_{k-1}, the time of the previous event or measurement, forward to time t_j using the prediction equations found in the NAVM Analysis section.

For any event other than a prediction event, subroutine SETEVN also computes eigenvalues, eigenvectors, and hyperellipsoids of the position and velocity partitions of the knowledge covariance at t_j.

SETEVN Flow Chart



SUBROUTINE SETEVS

PURPOSE PERFORM ALL COMPUTATIONS COMMON TO MOST EVENTS IN THE SIMULATION PROGRAM

CALLING SEQUENCE: CALL SETEVS(RI,TEVN,RI1,NCODE)

ARGUMENT:	NCODE	I	EVENT CODE
	RI	I	TARGETED NOMINAL SPACECRAFT STATE AT PREVIOUS MEASUREMENT OR EVENT TIME
	RI1	I	MOST RECENT NOMINAL SPACECRAFT STATE AT PREVIOUS MEASUREMENT OR EVENT TIME
	TEVN	I	EVENT TIME

SUBROUTINES SUPPORTED: SIMULL

SUBROUTINES REQUIRED: CORREL DYNOS HYELS JACOBI NAVM NTMS PSIM STMPR TITLES TRAPAR

LOCAL SYMBOLS	DUM	INTERMEDIATE VECTOR
	EGVT	ARRAY OF EIGENVECTORS
	EGVL	ARRAY OF EIGENVALUES
	ICODE	INTERNAL CONTROL FLAG
	OUT	SQUARE ROOTS OF EIGENVALUES
	PEIG	MATRIX WHOSE HYPERELLIPSOID IS TO BE COMPUTED
	RF1	MOST RECENT NOMINAL SPACECRAFT STATE AT EVENT TIME
	RF	TARGETED NOMINAL SPACECRAFT STATE AT EVENT TIME
	VEIG	MATRIX TO BE DIAGONALIZED

COMMON COMPUTED/USED: ADEVX EDEVX TRTM1 XF1 XF XI1 ZF ZI

COMMON COMPUTED: DELTM XI

COMMON USED: ADEVXS CXSU CXSV CXU CXV CXXS EDEVXS FOP FOV IEIG IHYP1 IPRT ISTMC NDIM1 NGE

NQE	NTMC	PHI	PS	P
Q	TXXS	U0	V0	W
XLAB				

SETEVS Analysis

Prior to executing any event in the simulation mode, subroutine SETEVS is called to perform a series of computations which are common to all events. After computing the targeted nominal and most recent nominal states at the time of the event t_j, knowledge covariance partitions are propagated forward to time t_j from time t_{k-1} of the previous event or measurement using the prediction equations found in the NAVM Analysis section. The actual trajectory state at t_j is computed using

$$X_j = Z_j + \omega_j$$

where Z_j is the actual trajectory state assuming no unmodeled acceleration has been acting on the spacecraft, and ω_j is the contribution of the actual unmodeled acceleration to the actual trajectory state at t_j. The actual and predicted position/velocity deviations from the most recent nominal at t_j are given by

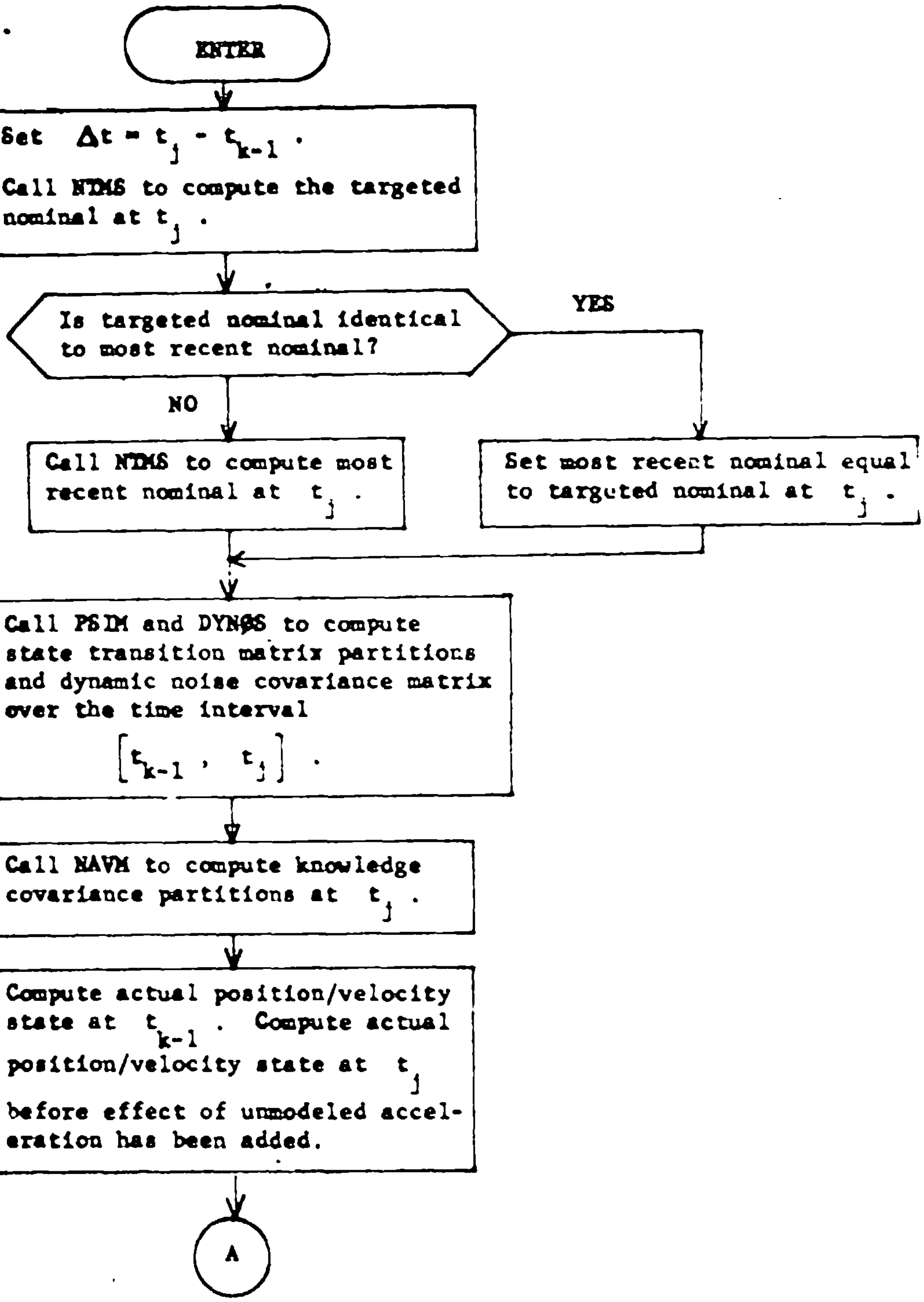
$$\delta\tilde{X}_j = X_j - \tilde{X}_j$$

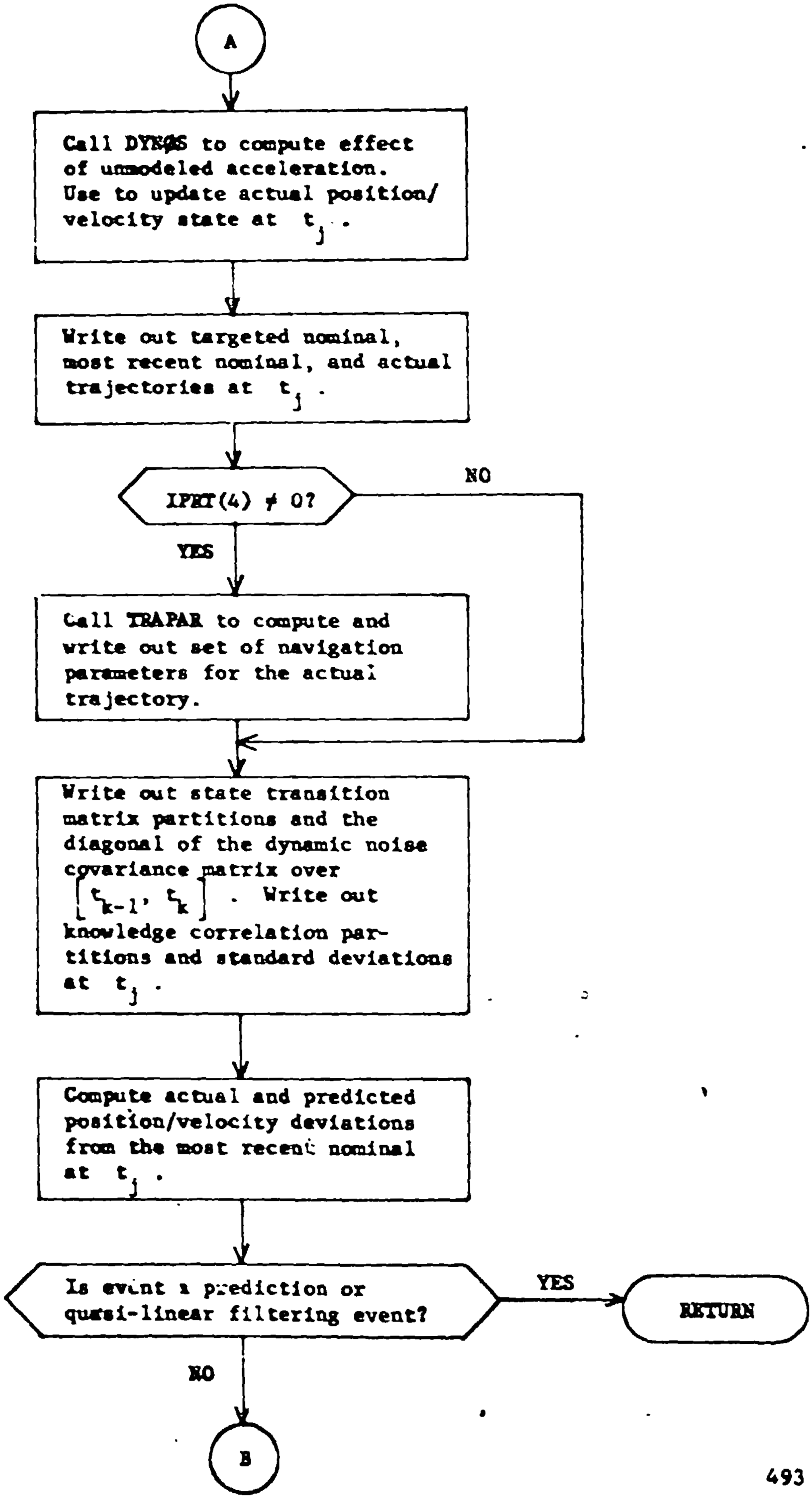
and $$\delta\tilde{\hat{X}}_j = \Phi(t_j, t_{k-1})\, \delta\tilde{\hat{X}}_{k-1} + \theta_{xx_s}(t_j, t_{k-1})\, \delta\tilde{\hat{X}}_{s_j} ,$$

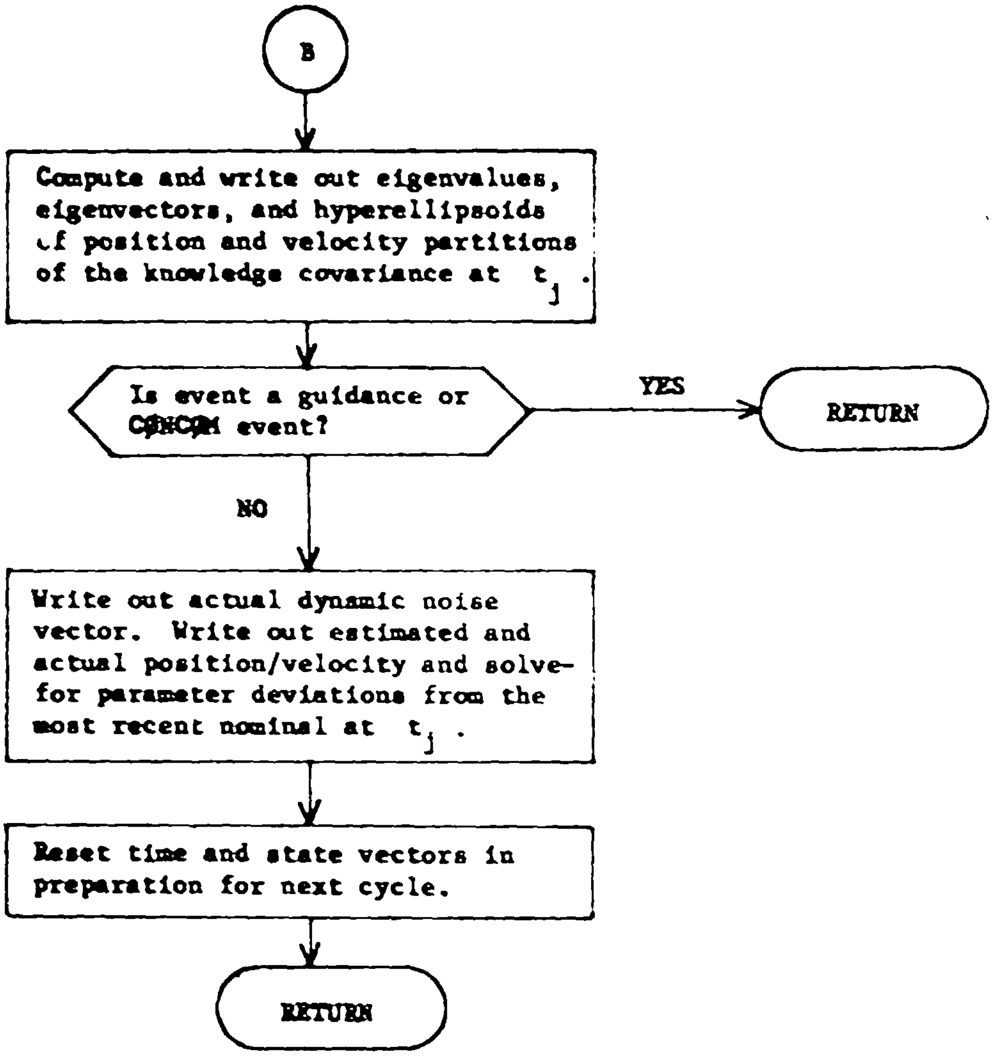
respectively, where Φ and θ_{xx_s} are the state transition matrix partitions over $\left[t_{k-1}, t_j \right]$.

For any event other than prediction and quasi-linear filtering events, subroutines SETEVS also computes eigenvalues, eigenvectors, and hyperellipsoids of the position and velocity partitions of the knowledge covariance at t_j.

SETEVS Flow Chart




A
Call DYNØS to compute effect of unmodeled acceleration. Use to update actual position/velocity state at t_j .
Write out targeted nominal, most recent nominal, and actual trajectories at t_j .
IPRT(4) ≠ 0?
NO
YES
Call TRAPAR to compute and write out set of navigation parameters for the actual trajectory.
Write out state transition matrix partitions and the diagonal of the dynamic noise covariance matrix over $[t_{k-1}, t_k]$. Write out knowledge correlation partitions and standard deviations at t_j .
Compute actual and predicted position/velocity deviations from the most recent nominal at t_j .
Is event a prediction or quasi-linear filtering event?
YES
RETURN
NO
B


B
Compute and write out eigenvalues, eigenvectors, and hyperellipsoids of position and velocity partitions of the knowledge covariance at t_j .
Is event a guidance or CONCOM event?
YES
RETURN
NO
Write out actual dynamic noise vector. Write out estimated and actual position/velocity and solve-for parameter deviations from the most recent nominal at t_j .
Reset time and state vectors in preparation for next cycle.
RETURN

PROGRAM SIMUL

PURPOSE: TO CONTROL THE COMPUTATIONAL FLOW THROUGH THE BASIC CYCLE (MEASUREMENT PROCESSING) AND ALL EVENTS IN THE SIMULATION PROGRAM

SUBROUTINES SUPPORTED: MAIN

SUBROUTINES REQUIRED: BIAS DYNOS MENOS NAVM NTMS TRAKS PRINT4 PSIM SCHED SETEVS

LOCAL SYMBOLS

BVAL	ACTUAL MEASUREMENT BIAS VECTOR
DUMM	INTERMEDIATE VARIABLE
DUM	INTERMEDIATE VECTOR
IPRN	MEASUREMENT PRINT TIME COUNTER
MMCODE	MEASUREMENT CODE
NEVENT	EVENT COUNTER
RNUM	RANDOM MEASUREMENT NOISE
TRTM2	TIME OF THE MEASUREMENT

COMMON COMPUTED/USED: ADEVX ANGIS AY EDEVXS EDEVX EY ICODE MCNTR NAFC RES RF1 RI1 RI TEVN TRTM1 XF1 XF XI1 XI ZF ZI

COMMON COMPUTED: AYMEY DELTM EDEVSM EDEVXM

COMMON USED: AK AM AR FNTM H IEVNT IPRINT ISTMC NAE NAF6 NDIM1 NEV NGE NMN NQE NR NTMC PHI RF S TEV TXXS W ZERO

SIMUL Analysis

The primary function of subroutine SIMUL is to control the computational flow through the basic cycle (measurement processing) and all events in the simulation mode. Subroutine SIMUL also performs some computations in the basic cycle. All event- elated analysis is presented in the event subroutines themselves and will not be treated below.

In the basic cycle the first task of SIMUL is to control the generation of targeted nominal and most recent nominal spacecraft states, $\bar{X}_{k+1}$ and $\tilde{X}_{k+1}$, respectively, at time t_k, given states $\bar{X}_k$ and $\tilde{X}_k$ at time t_k. Then, calling PSIM, DYNØS, TRAKS, and MENØS, successively, SIMUL controls the computation of all matrix information required by subroutine NAVM in order to compute the covariance matrix partitions at time t_{k+1}^{+} immediately following the measurement.

After computing the actual state X_k at time t_k from

$$X_k = \tilde{X}_k + \delta\tilde{X}_k \qquad (1)$$

where $\delta\tilde{X}_k$ is the actual spacecraft state deviation from the most recent nomin.., SIMUL controls the generation of the actual state Z_{k+1} at time t_k before the effect of unmodeled acceleration has been added. Then, having called DYNØS to compute the effect of unmodeled acceleration ω_{k+1}, SIMUL computes the actual state and actual state deviation at time t_{k+1}:

$$X_{k+1} = Z_{k+1} + \omega_{k+1} \qquad (2)$$

$$\delta\tilde{X}_{k+1} = X_{k+1} - \tilde{X}_{k+1} \qquad (3)$$

With both the most recent nominal and actual spacecraft states available at t_{k+1}, SIMUL calls TRAKS twice in succession to compute the ideal measurements $\tilde{Y}_{k+1}$ and $\underline{Y}_{k+1}$, respectively, which would be made at each of these trajectory states. Calling MENØS, RNUM, and BIAS to compute the actual measurement noise and bias corrupting the ideal measurement associated with the actual state, SIMUL computes the actual measurement at time t_{k+1} using

$$Y^a_{k+1} = \underline{Y}_{k+1} + b_{k+1} + \nu_{k+1} \qquad (4)$$

where b_{k+1} and ν_{k+1} represent the actual measurement bias and noise, respectively.

All information required for computing both predicted and filtered state deviations from the most recent nominal at t_{k+1} is now available. With Φ and Θ_{xx_s} denoting state transition matrix partitions over the time interval $\left[t_k, t_{k+1} \right]$, SIMUL computes the predicted spacecraft state deviations and solve-for parameter deviations at t_{k+1} using

$$\delta\tilde{\hat{X}}^-_{k+1} = \Phi\,\delta\tilde{\hat{X}}^+_k + \Theta_{xx_s}\,\delta\tilde{\hat{X}}^+_{s_k} \tag{5}$$

$$\delta\tilde{\hat{X}}^-_{s_{k+1}} = \delta\tilde{\hat{X}}^+_{s_k} \tag{6}$$

Prior to computing filtered deviations, SIMUL computes the measurement residual from

$$\epsilon_{k+1} = (Y^a_{k+1} - \tilde{Y}_{k+1}) - H_{k+1}\,\delta\tilde{\hat{X}}^-_{k+1} - M_{k+1}\,\delta\tilde{\hat{X}}^-_{s_{k+1}} \tag{7}$$

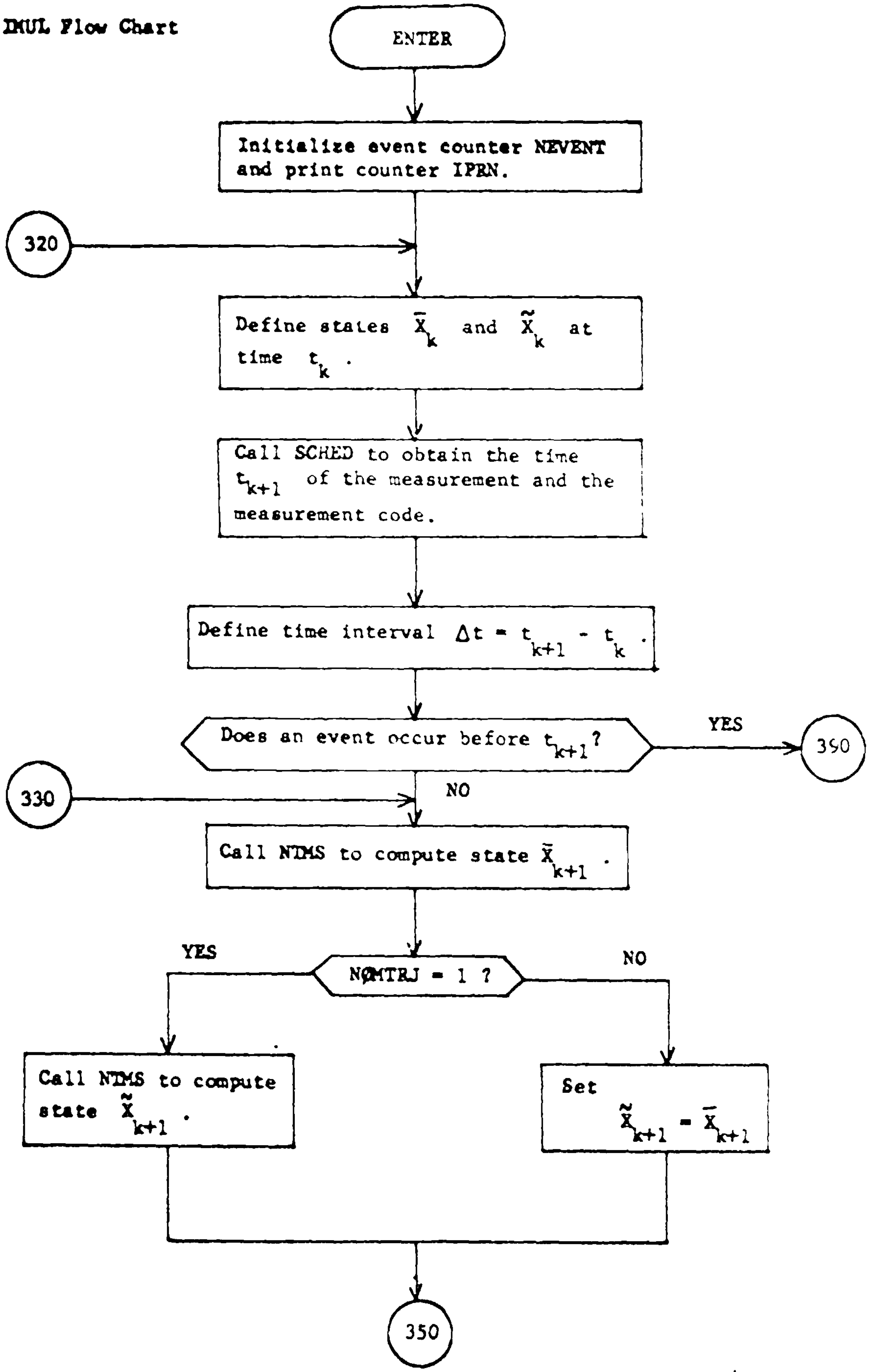
where H_{k+1} and M_{k+1} are observation matrix partitions. Filtered spacecraft state deviations and solve-for parameter deviations are then computed from

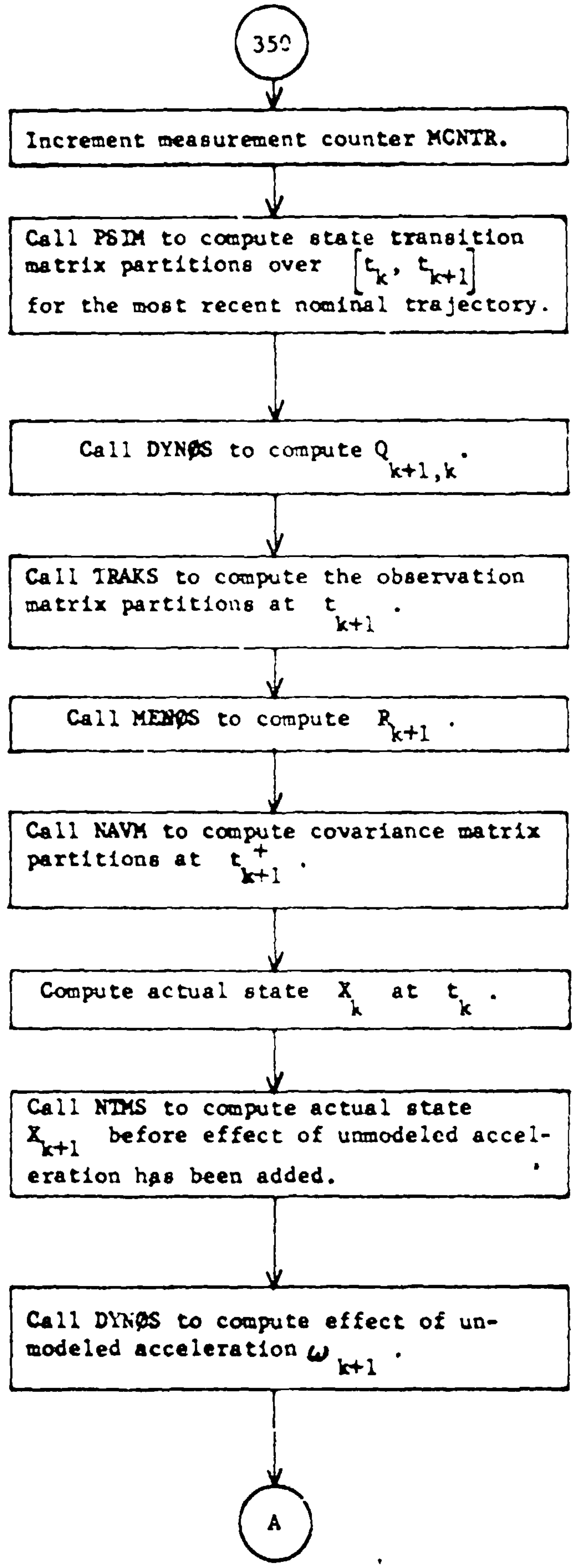
$$\delta\tilde{\hat{X}}^+_{k+1} = \delta\tilde{\hat{X}}^-_{k+1} + K_{k+1}\,\epsilon_{k+1} \tag{8}$$

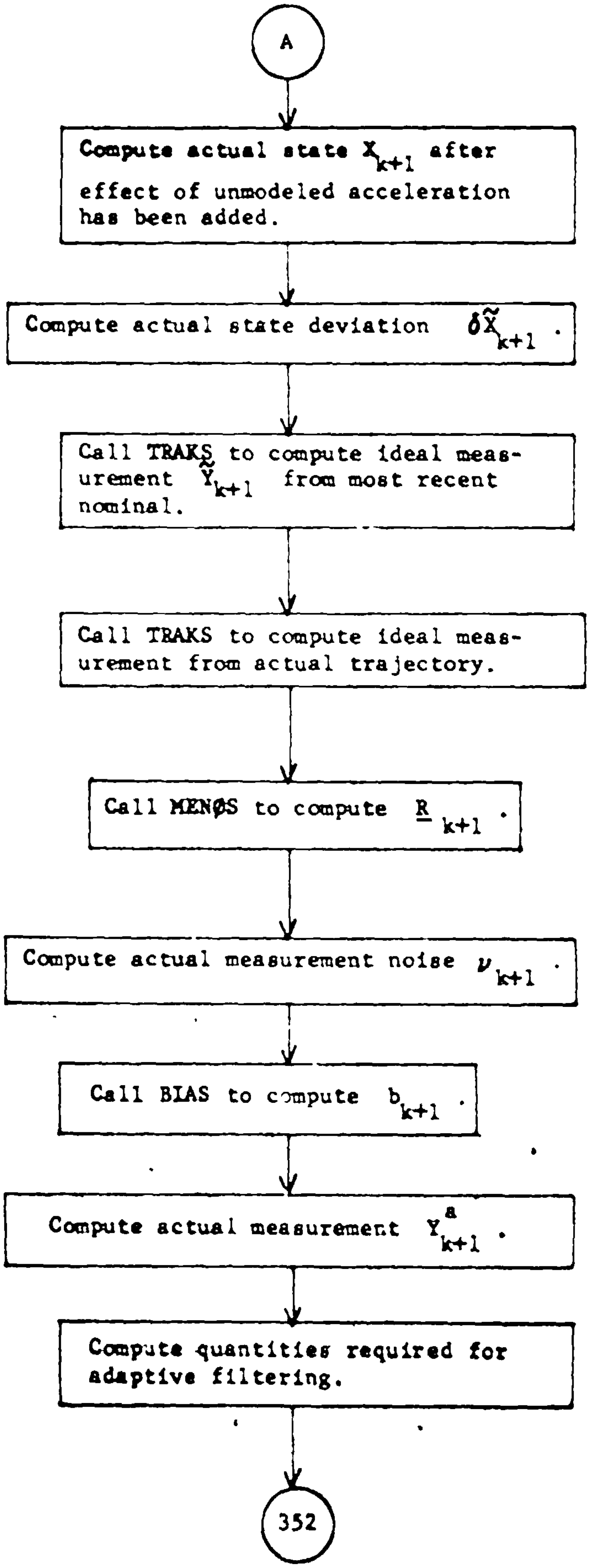
$$\delta\tilde{\hat{X}}^+_{s_{k+1}} = \delta\tilde{\hat{X}}^-_{s_{k+1}} + S_{k+1}\,\epsilon_{k+1} \tag{9}$$

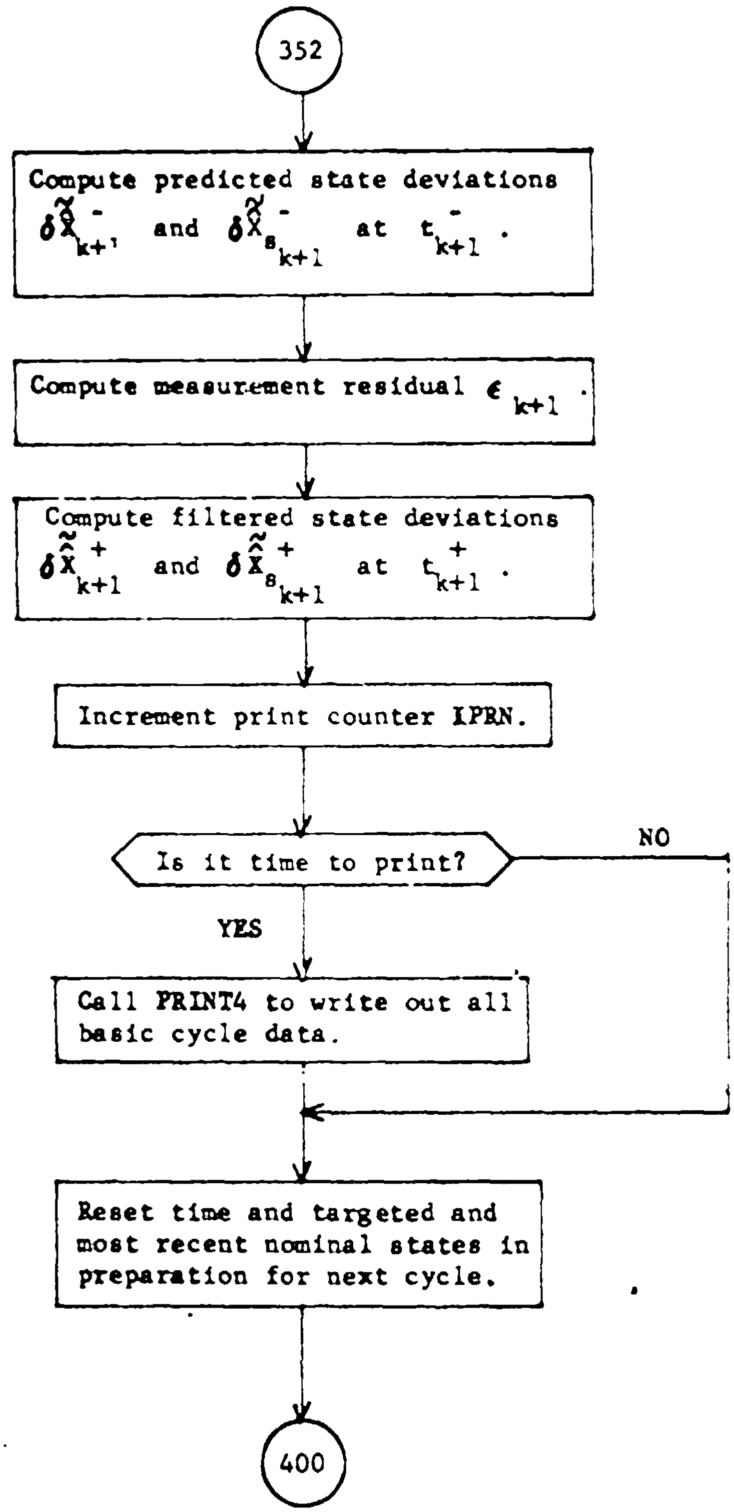
where K_{k+1} and S_{k+1} are the filter gain constants.

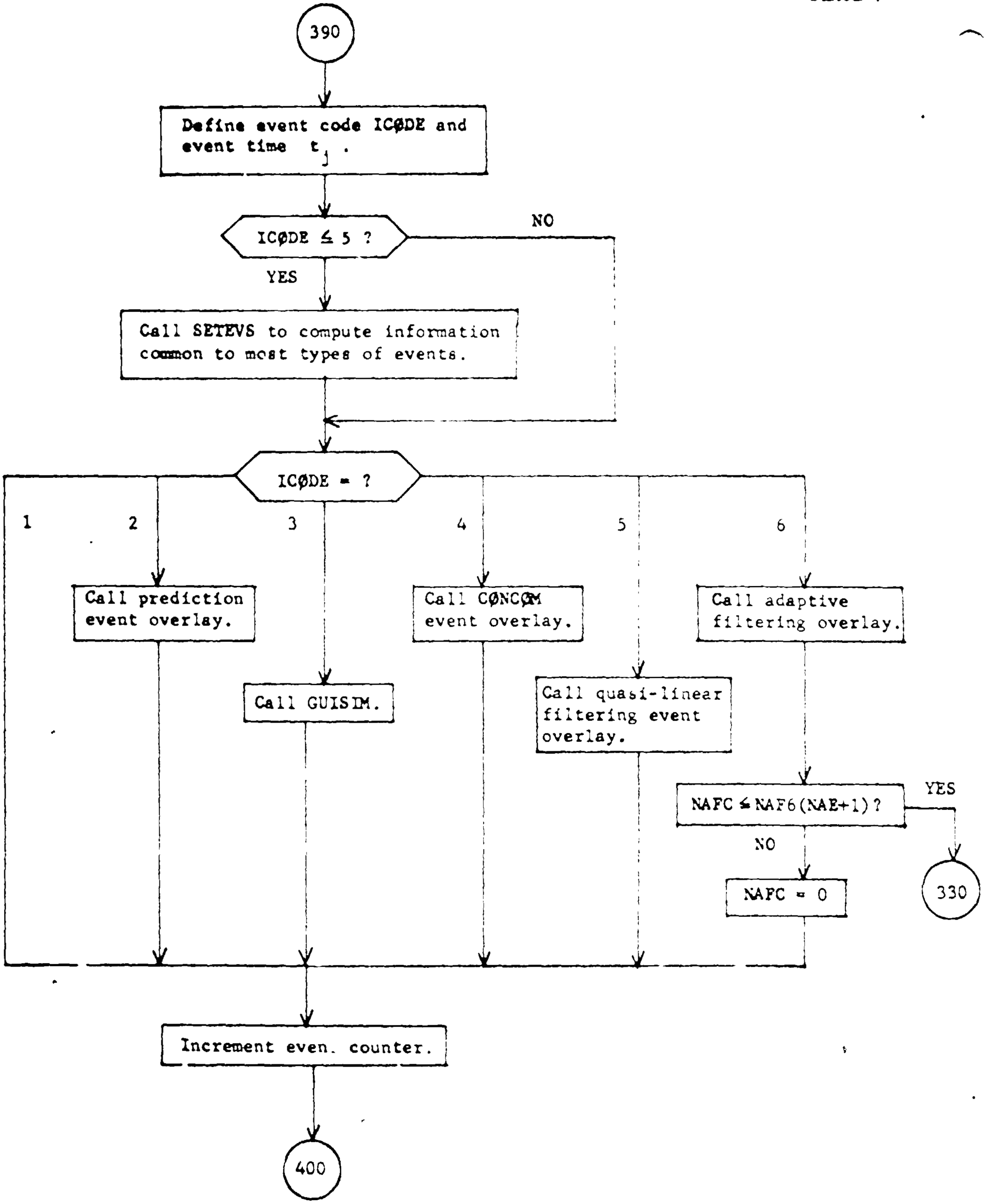
SIMUL Flow Chart

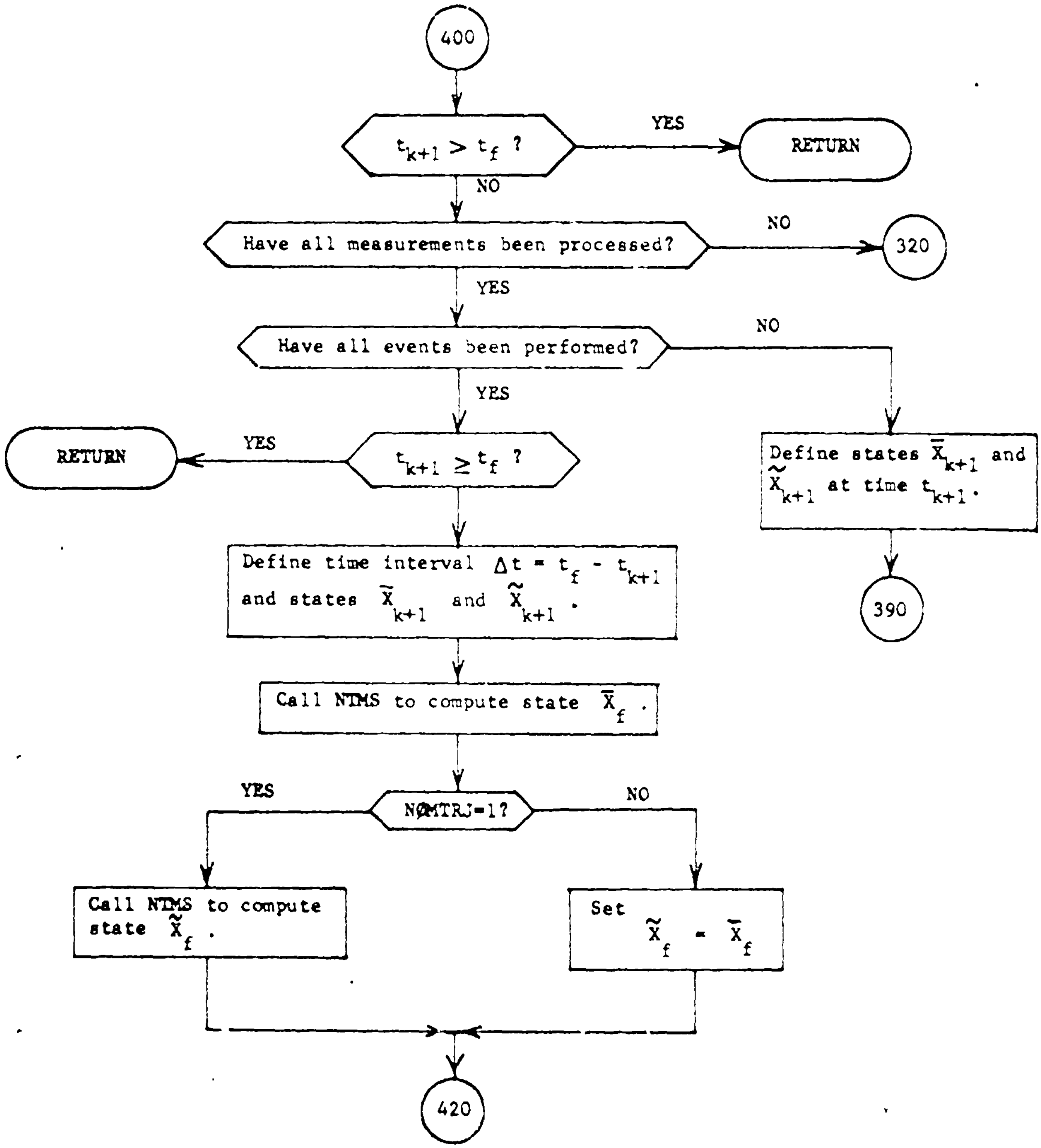


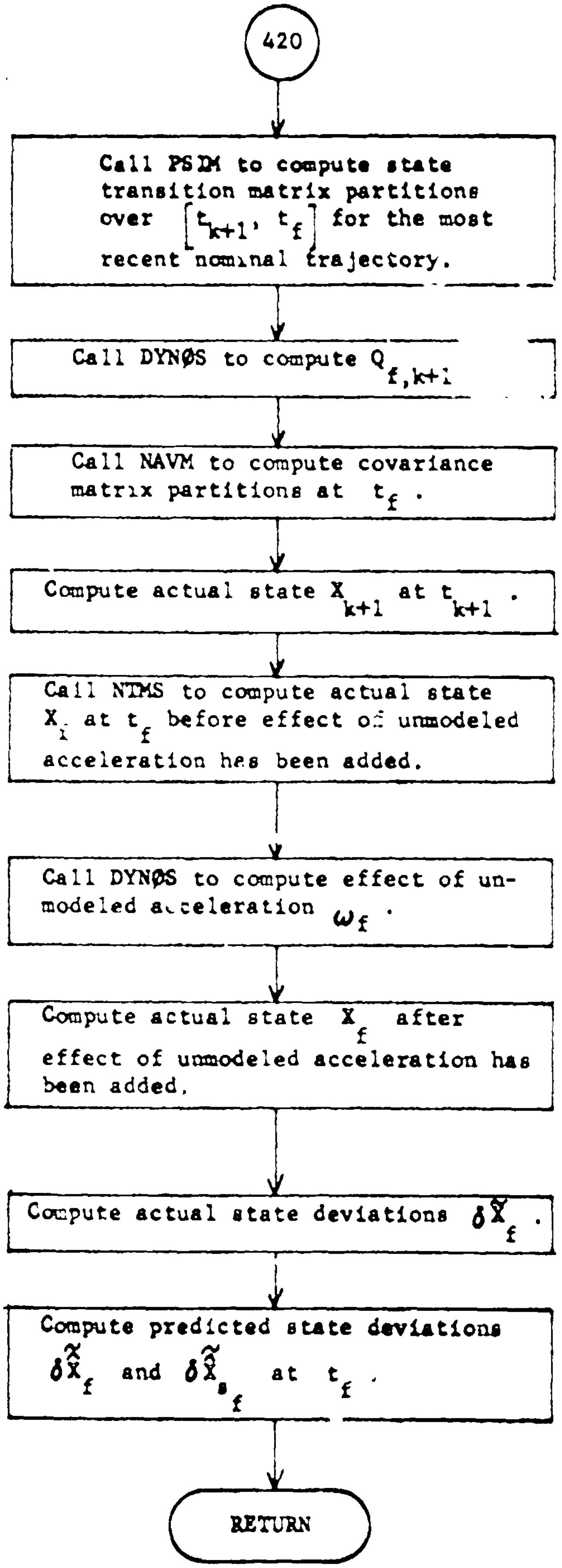

350
Increment measurement counter MCNTR.
Call PSIM to compute state transition matrix partitions over $[t_k, t_{k+1}]$ for the most recent nominal trajectory.
Call DYNØS to compute $Q_{k+1,k}$.
Call TRAKS to compute the observation matrix partitions at t_{k+1}.
Call MENØS to compute R_{k+1}.
Call NAVM to compute covariance matrix partitions at t_{k+1}^{+}.
Compute actual state X_k at t_k.
Call NTMS to compute actual state X_{k+1} before effect of unmodeled acceleration has been added.
Call DYNØS to compute effect of unmodeled acceleration ω_{k+1}.
A


A
Compute actual state X_{k+1} after effect of unmodeled acceleration has been added.
Compute actual state deviation $\delta\tilde{X}_{k+1}$.
Call TRAKS to compute ideal measurement $\tilde{Y}_{k+1}$ from most recent nominal.
Call TRAKS to compute ideal measurement from actual trajectory.
Call MENØS to compute $\underline{R}_{k+1}$.
Compute actual measurement noise ν_{k+1} .
Call BIAS to compute b_{k+1} .
Compute actual measurement Y^{a}_{k+1} .
Compute quantities required for adaptive filtering.
352


352
Compute predicted state deviations $\delta\tilde{x}^{-}_{k+1}$ and $\delta\tilde{x}^{-}_{s_{k+1}}$ at t^{-}_{k+1}.
Compute measurement residual ϵ_{k+1}.
Compute filtered state deviations $\delta\hat{\tilde{x}}^{+}_{k+1}$ and $\delta\hat{\tilde{x}}^{+}_{s_{k+1}}$ at t^{+}_{k+1}.
Increment print counter IPRN.
Is it time to print?
NO
YES
Call PRINT4 to write out all basic cycle data.
Reset time and targeted and most recent nominal states in preparation for next cycle.
400


390
Define event code ICØDE and event time t_j .
ICØDE ≤ 5 ?
NO
YES
Call SETEVS to compute information common to most types of events.
ICØDE = ?
1
2
3
4
5
6
Call prediction event overlay.
Call CØNCØM event overlay.
Call adaptive filtering overlay.
Call GUISIM.
Call quasi-linear filtering event overlay.
NAFC ≤ NAF6(NAE+1) ?
YES
NO
NAFC = 0
330
Increment even. counter.
400


400
$t_{k+1} > t_f$?
YES
RETURN
NO
Have all measurements been processed?
NO
320
YES
Have all events been performed?
NO
YES
RETURN
YES
$t_{k+1} \geq t_f$?
Define states $\bar{X}_{k+1}$ and $\tilde{X}_{k+1}$ at time t_{k+1}.
390
Define time interval $\Delta t = t_f - t_{k+1}$ and states $\bar{X}_{k+1}$ and $\tilde{X}_{k+1}$.
Call NTMS to compute state $\bar{X}_f$.
YES
NØMTRJ=1?
NO
Call NTMS to compute state $\tilde{X}_f$.
Set $\tilde{X}_f = \bar{X}_f$
420


420
Call PSIM to compute state transition matrix partitions over $[t_{k+1}, t_f]$ for the most recent nominal trajectory.
Call DYNØS to compute $Q_{f,k+1}$
Call NAVM to compute covariance matrix partitions at t_f.
Compute actual state X_{k+1} at t_{k+1}.
Call NTMS to compute actual state X_i at t_f before effect of unmodeled acceleration has been added.
Call DYNØS to compute effect of unmodeled acceleration ω_f.
Compute actual state X_f after effect of unmodeled acceleration has been added.
Compute actual state deviations $\delta \tilde{X}_f$.
Compute predicted state deviations $\delta \hat{\tilde{X}}_f$ and $\delta \hat{\tilde{X}}_{s_f}$ at t_f.
RETURN

SUBROUTINE SPACE

PURPOSE: COUNTS THE NUMBER OF LINES BEING PRINTED TO DETERMINE WHEN TO SKIP TO THE NEXT PAGE WITH A NEW HEADING

CALLING SEQUENCE CALL SPACE(LINES)

ARGUMENT LINES I NUMBER OF LINES THAT WILL BE WRITTEN IN THE NEXT OUTPUT STATEMENT

SUBROUTINES SUPPORTED: INPUTZ PRINT VECTOR VMP

SUBROUTINES REQUIRED: NEWPGE

COMMON COMPUTED/USED: LINCT

COMMON USED: LINPGE

SUBROUTINE STAPRL

PURPOSE: TO COMPUTE THE PARTIAL DERIVATIVES OF STATION LOCATION ERRORS.

CALLING SEQUENCE: CALL STAPRL(AL,ALON,ALAT,PAT2,VEC,PA)

ARGUMENT:	AL	I	ALTITUDE OF THE STATION
	ALAT	I	LATITUDE OF THE STATION
	ALON	I	LONGITUDE OF THE STATION
	PA	O	PARTIAL OF STATION POSITION AND VELOCITY WITH RESPECT TO ALTITUDE, LATITUDE AND LONGITUDE
	PAT2	I	LONGITUDE + OMEGA*(CURRENT TIME-LAUNCH TIME)
	VEC		UNUSED

SUBROUTINES SUPPORTED: TRAKS TRAKM

LOCAL SYMBOLS:	G1	SINE OF LATITUDE
	G2	COSINE OF LATITUDE
	G3	SINE(PHI +OMEGA(T-UNIVT))
	G4	COSINE(PHI +OMEGA(T-UNIVT)) WHERE PHI =LONGITUDE OMEGA=EARTH ROTATION RATE T =TIME UNIVT=UNIVERSAL TIME
	G5	SINE OF OBLIQUITY OF EARTH
	G6	COSINE OF OBLIQUITY OF EARTH
	OMEG	OMEGA IN PROPER UNITS

COMMON USED: [illegible]S OMEGA TM

STAPRL Analysis

The ecliptic components of the position and velocity of a tracking station relative to the Earth are related to station location parameters R, θ, and ϕ through the following set of equations:

$$X_s = R \cos\theta \cos G$$

$$Y_s = R \cos\theta \cos\epsilon \sin G + R \sin\theta \sin\epsilon$$

$$Z_s = -R \cos\theta \sin\epsilon \sin G - R \sin\theta \cos\epsilon$$

$$\dot{X}_s = -\omega R \cos\theta \sin G$$

$$\dot{Y}_s = \omega R \cos\theta \cos\epsilon \cos G$$

$$\dot{Z}_s = -\omega R \cos\theta \sin\epsilon \cos G$$

where $G = \phi + \omega(t - T)$, and T is the universal time at some epoch(usually launch time).

Subroutine STAPRL computes the negative of the partials of the previous quantities with respect to the station location parameters R, θ, and ϕ. These partials are summarized below:

$$-\frac{\partial X_s}{\partial R} = -\cos\theta \cos G$$

$$-\frac{\partial X_s}{\partial \theta} = R \sin\theta \cos G$$

$$-\frac{\partial X_s}{\partial \phi} = R \cos\theta \sin G$$

$$-\frac{\partial Y_s}{\partial R} = -\left[\sin\epsilon \sin\theta + \cos\epsilon \cos\theta \sin G\right]$$

$$-\frac{\partial Y_s}{\partial \theta} = R \cos\epsilon \sin\theta \sin G - R \sin\epsilon \cos\theta$$

$$-\frac{\partial Y_s}{\partial \phi} = -R \cos\epsilon \cos\theta \cos G$$

$$-\frac{\partial Z_s}{\partial R} = \sin\epsilon \cos\theta \sin G - \cos\epsilon \sin\theta$$

$$-\frac{\partial Z_s}{\partial \theta} = -\left[R \sin \epsilon \sin \theta \sin G + R \cos \epsilon \cos \theta\right]$$

$$-\frac{\partial Z_s}{\partial \phi} = R \sin \epsilon \cos \theta \cos G$$

$$-\frac{\partial \dot{X}_s}{\partial R} = \omega \cos \theta \sin G$$

$$-\frac{\partial \dot{X}_s}{\partial \theta} = -\omega R \sin \theta \sin G$$

$$-\frac{\partial \dot{X}_s}{\partial \phi} = \omega R \cos \theta \cos G$$

$$-\frac{\partial \dot{Y}_s}{\partial R} = -\omega \cos \theta \cos \epsilon \cos G$$

$$-\frac{\partial \dot{Y}_s}{\partial \theta} = \omega R \cos \epsilon \sin \theta \cos G$$

$$-\frac{\partial \dot{Y}_s}{\partial \phi} = \omega R \cos \epsilon \cos \theta \sin G$$

$$-\frac{\partial \dot{Z}_s}{\partial R} = \omega \sin \epsilon \cos \theta \cos G$$

$$-\frac{\partial \dot{Z}_s}{\partial \theta} = -\omega R \sin \epsilon \sin \theta \cos G$$

$$-\frac{\partial \dot{Z}_s}{\partial \phi} = -\omega R \sin \epsilon \cos \theta \sin G$$

SUBROUTINE STMPR

PURPOSE: TO PRINT OUT THE TRANSPOSES OF THE STATE TRANSITION MATRIX PARTITIONS PHI, TXXS, AND TXU OVER AN ARBITRAR INTERVAL OF TIME.

CALLING SEQUENCE: CALL STMPR(TRTM1,TRTM2)

ARGUMENT:	TRTM1	I	TIME AT BEGINNING OF INTERVAL OVER WHICH STATE TRANSITION MATRIX PARTITIONS HAVE BEEN COMPUTED
	TRTM2	I	TIME AT END OF INTERVAL OVER WHICH STATE TRANSITION MATRIX PARTITIONS HAVE BEEN COMPUTED

SUBROUTINES SUPPORTED: PRINT4 SETEVS GUISIM GUISS PRESIM
PRINT3 SETEVN GUIDM GUID PRED

COMMON USED: NDIM1 NDIM2 PHI TXU TXXS
XLAB XSL XU

SUBROUTINE SUB1

PURPOSE: TO COMPUTE POSITION AND VELOCITY MAGNITUDES.

CALLING SEQUENCE: CALL SUB1(X,XE,XP)

ARGUMENT:	X	I	INERTIAL POSITION/VELOCITY OF THE VEHICLE
	XE	I	EARTH-S POSITION/VELOCITY
	XP	I	POSITION/VELOCITY OF THE TARGET PLANET

SUBROUTINES SUPPORTED: PRINT4

LOCAL SYMBOLS:	RX	MAGNITUDE OF INERTIAL POSITION VECTOR
	RY	MAGNITUDE OF GEOCENTRIC POSITION VECTOR
	RZ	MAGNITUDE OF PLANETOCENTRIC POSITION VECTOR
	VX	MAGNITUDE OF INERTIAL VELOCITY VECTOR
	VY	MAGNITUDE OF GEOCENTRIC VELOCITY VECTOR
	VZ	MAGNITUDE OF PLANETOCENTRIC VELOCITY VECTOR
	Y	GEOCENTRIC POSITION/VELOCITY OF THE VEHICLE
	Z	PLANETOCENTRIC POSITION/VELOCITY OF THE VEHICLE

SUBROUTINE TARGET

PURPOSE: TO PERFORM EXECUTIVE FUNCTIONS OF THE TARGETING MODE AS CALLING REQUIRED SUBROUTINES TO READ THE INPUT DATA, COMPUTING THE ZERO ITERATE IF NECESSARY AND PERFORMING THE ACTUAL TARGETING THROUGH THE PROGRESSIVE STAGES USED BY STEAP.

CALLING SEQUENCE: CALL TARGET

SUBROUTINES SUPPORTED: GIDANS

SUBROUTINES REQUIRED: TAROPT TARMAX DESENT PECEQ VMP

LOCAL SYMBOLS:

ABV	INTERMEDIATE VARIABLE USED TO LIMIT EACH DELTAV COMPONENT CHANGE
ACC	VECTOR OF ACCURACY LEVELS FOR THE CURRENT TARGETING EVENT
ACK	ACTUAL ACCURACY USED BY SUBROUTINE VMP
AER	ABSOLUTE VALUES OF DIFFERENCES BETWEEN DESIRED AND NOMINAL END CONDITIONS
CERROR	CURRENT SUM OF WEIGHTED DIFFERENCES OF DESIRED AUXILIARY AND NOMINAL AUXILIARY END CONDITIONS
DEV	DIFFERENCES (ERRORS) BETWEEN AUXILIARY END CONDITIONS (DESIRED AND NOMINAL)
ISP2	INDICATOR USED BY SUBROUTINE VMP =1 STOP AT SPHERE-OF-INFLUENCE =0 DO NO STOP AT SPHERE-OF-INFLUENCE
ITBAD	BAD STEP COUNTER
ITER	ITERATION COUNTER
ITOL	CONVERGENCE INDICATOR =1 CASE CONVERGED =0 CASE DID NOT CONVERGE
IT	INDICATOR USED TO LOCATE DESIRED TIME VALUE FOR OUTER TARGETING
I	INDEX
J	INDEX
LOWHI	INDICATOR USED TO CALCULATE THE PHASE 2

TARGETING MATRIX

NOMORE	INDICATOR USED TO LIMIT OUTER TARGETING =0 OUTER TARGETING HAS NOT BEEN PERFORMED =1 OUTER TARGETING HAS ALREADY BEEN PERFORMED
OSPH	ORIGINAL SPHER OF INFLUENCE OF THE TARGET PLANET
PERROR	PREVIOUS VALUE OF CERROR
REDUC	INTERMEDIATE VARIABLE USED IN BAD STEP REDUCTION
RIS	LOCAL VECTOR USED TO SAVE AND RESTORE THE RIN VECTOR
RR	INTERMEDIATE VARIABLE FOR OUTER TARGETING
RSF	FINAL SPACECRAFT STATE RETURNED BY VMP
SSOI	INTERMEDIATE VARIABLE FOR OUTER TARGETING
STOL	INTERMEDIATE VARIABLE FOR OUTER TARGETING
TMDF	INTERMEDIATE VARIABLE FOR OUTER TARGETING
TVH	PHASE 1 TARGETED VELOCITY AT HIGHEST ACCURACY
TVL	PHASE 1 TARGETED VELOCITY AT LOSEST ACCURACY
VV	INTERMEDIATE VARIABLE FOR OUTER TARGETING
XTIME	CURRENT DT TIME USED TO CALCULATE EQECP FOR TARGET PLANET

COMMON COMPUTED/USED:	CTOL	DAUX	DELTAV	DTAR	IBAD
	IBAST	IPHASE	ISPH	ISTART	ISTOP
	ITARM	KEYTAR	LEVELS	LEV	MATX
	MAXBAD	NITS	NOPAR	NOPHAS	NOSOI
	PHI	RIN	SPHERE		
COMMON COMPUTED:	DELTP	DELV	ICL2	ICL	INCMT
	INPR	IPRINT	XAXTAR	KNIT	RRF

```
COMMON USED:            AAUX    AC      ALNGTH  ATAR    DC
                        DELTAT  DT      DVMAX   D1      EQECP
                        FAC     IBADS   KTAR    KUR     LVLS
                        MAT     MAXB    NOIT    NPAR    NTP
                        ONE     RC      SPHFAC  TAR     TM
                        TOL     TRTM    TWO     ZERO
```

TARGET Analysis

TARGET is responsible for the control of any targeting (nonlinear guidance) event. The targeting is done either by the Newton-Raphson technique or by a steepest descent-conjugate gradient algorithm, the method being specified by the user. In either case numerical differencing is used to compute the required sensitivities.

I. Preliminaries

The current inertial state of the spacecraft upon entering TARGET is first saved (RIS=RIN) along with the original SOI radius (OSPH=SPHERE) since both variables may be changed during the course of the targeting. Before exiting from TARGET these values are restored.

The index of the current event KUR has been computed by TRJTRY. This enables the specific targeting parameters for the current event to be set:

Parameter	Definition
METHOD	Triggers Newton Raphson (=0) or Steepest Descent (≠0) technique
MATX	Determines whether Newton-Raphson matrix is computed always (=2) or only at low level (=1)
IBAST	Determines whether bad step checks are made never (=1), high level only (=2) or always (=3)
LEVELS	Number of integration accuracy levels to be used
NOPAR	Number of target parameters to be used
ACC	Actual accuracy levels used in targeting

The following flags are then initialized to zero

Flag	Definition
ITDS	Counter for steepest descent iterations
LOWHI	Flag indicating whether first phase complete (=1) or not (=0)
NOMORE	Flag indicating whether outer targeting has been done (=1) or not (=0)

The target time is computed and using that time the transformation matrix ϕ_{ECEQ} from ecliptic to target planet equatorial coordinates is calculated (PECEQ).

II. Phase Preparations

TARGET performs the targeting in one phase unless targeting to TCA (time of closest approach). In that case the trajectory is targeted in two phases: the first phase targets to the target planet SOI (sphere of influence), the second phase to the closest approach conditions. IPHASE is the phase counter, NOPHAS is the number of phases needed.

If all the phases have been completed, the program prepares to exit. If the last iterate satisfied the target tolerances ITOL will have been set to a 1. If it did not, ITOL will be zero and this requires that KWIT be set to 1 to terminate the program upon return to the basic cycle.

If the last phase has not yet been completed TAROPT is now called with an argument 1 to compute the following phase parameters:

Parameter	Definition
KEYTAR(3)	Vector of codes of target parameters
KAXTAR(3)	Vector of codes of auxiliary parameters
DTAR(3)	Vector of desired values of target parameters
DAUX(3)	Vector of desired values of auxiliary parameters
FAC(3)	Weighting factors for loss function for auxiliary parameters
ISTOP	Flag indicating integration stopping conditions with ISTOP = 1,2,3 indicating fixed final time, SOI, or CA encounter

The target parameters are the parameters actually desired; the auxiliary parameters are the parameters used to do the targeting. The target and auxiliary parameters are identical except when i_{CA} and r_{CA} are targets. In that case the corresponding auxiliary parameters are $B \cdot T$ and $B \cdot R$ which are much more linear variables. The codes of the target and auxiliary parameters are as follows:

Code	1	2	3	4	5	6	7	8	9	10	11	12
Parameter	TRF*	TSI	TCS	TCA	BDT	BDR	RCA	INC	SMA	XRF	YRF	ZRF

* not currently available

III. Level Preparations

Within any phase TARGET operates through a series of integration accuracy levels prescribed by the user. After completing each level TARGET checks to see if the maximum number of levels LEVELS has been exceeded. If it has the program cycles to the beginning of the "phase loop" to go to the next phase. If the current level LEV is less than LEVELS the following computations are made.

The flag ITARM controls whether the previous targeting matrix is to be used (=1) or whether the matrix is to be recomputed (=2) during the current level. ITARM is set according to the current values of MATX, ISTART, and LEV.

The flag IBAD controls the bad step logic. If IBAD=1 no bad step check will be made during the current level; if IBAD=2 the bad step check will be in effect. TARGET sets IBAD according to the values of IBAST and LEV.

The flags ITOL, ITER, ITBAD are set to 0 to begin the iterations. The allowable iterations NITS and bad iterations MAXBAD are also set at this time.

IV. Iterate Calculations

Within each level the program makes one or more iterations. After each iteration the program updates the iteration counter ITER. If the maximum number of iterations for this level NITS has been exceeded, the program sets KWIT to 1 and prepares for the return from TARGET. Otherwise TARGET computes the target and auxiliary values corresponding to the current iterate values of state (position and velocity) RIN.

The integration parameters are first set. VMP is then called to propagate the initial state to the final stopping conditions. Checks are made to insure that the target planet SOI was intersected if the stopping conditions were SOI or CA. If it was not intersected and this is the first iteration, the "outer targeting" phase is entered (see below). If "outer targeting" has already been performed, the bad-step check is entered to reduce the previous correction by REDUC.

Otherwise TAROPT is called with the argument 2 to compute the desired and actual target (DTAR, ATAR) and auxiliary (DAUX, AAUX) parameter values. The absolute error in target values AER and the error in auxiliary values DEV are then computed.

If the current iterate is the first integration at the low level during the second phase of targeting (LOWHI=1) TARMAX is now called to compute the phase 2 targeting matrix. Then the state RIN is reset to the targeted velocity at the high level TVH to prepare for the second phase targeting. The program then returns to the level loop.

Otherwise the program now checks the actual target variables to determine whether they satisfy the input tolerances or not.

V. Tolerances Satisfied

If the tolerances are satisfied, the program first checks to see if the current targeting phase is outer targeting. If it is TARGET restores the original target parameters and initiates the normal targeting (see Outer Targeting below).

If the current targeting is already normal targeting, TARGET sets ITOL=1 to indicate the satisfaction of the tolerances. If the problem is a 2-phase and the current level is the highest level in phase 1 targeting, the targeted high level velocity TVH=RIN is saved, LOWHI is set to 1 and the targeted low level velocity is recalled RIN=TVL for the construct of the phase 2 targeting matrix. Then the level loop is reentered.

VI. Bad Step Reduction

If the target parameter values of any iterate are not within the acceptable tolerances TARGET now assigns a scalar error ϵ to the iterate using the weighting factors $\vec{W}$

$$\epsilon = \vec{W} \cdot \Delta \vec{T}$$

If the bad-step check is to be made on this iterate the current error ϵ is compared to the previous error ϵ_p. If $\epsilon \geq \epsilon_p$ and the maximum number of bad steps has not been exceeded, the previous correction $\Delta\vec{v}$ is reduced by REDUC (usually 1/4). The initial state RIN is adjusted by this and the iterate loop is reentered. If the maximum number of bad steps has been made, KWIT is set to 1 and the preparations for return are made.

VII. Generation of Next Iterate

The correction $\Delta\vec{v}$ to any iterate may be computed from either of two techniques selected by the flag METHOD. If METHOD $\neq$ 0, subroutine DESCENT is called for the computation of the $\Delta\vec{v}$ by a steepest descent algorithm. The numerical value of METHOD determines the number n of conjugate gradient steps between each straight gradient step where n = METHOD-1. Thus if METHOD=1, every step is in the direction of the gradient. But if METHOD=5, four steps are taken following the conjugate gradient direction before rectification by the gradient direction.

If METHOD=0, the Newton-Raphson correction is used. If ITARM=0, TARMAX is called for the computation of the targeting matrix ϕ by numerical differencing. If any of the integrations made in constructing that matrix satisfy the tolerances in τ, the flag IEND is set to 1 before returning to TARGET. Thus a check must be made on IEND. If ITARM=1 the previous targeting matrix is used. The correction is then given by

$$\Delta\vec{v} = \phi \cdot \Delta\vec{\alpha}$$

where $\Delta\vec{\alpha}$ are the deviations in the iterate auxiliary values. The $\Delta\vec{v}$ is checked to insure that the maximum step size DVMAX is not violated: if it is, the $\Delta\vec{v}$ is reduced proportionately to satisfy it. The next iterate is then set to

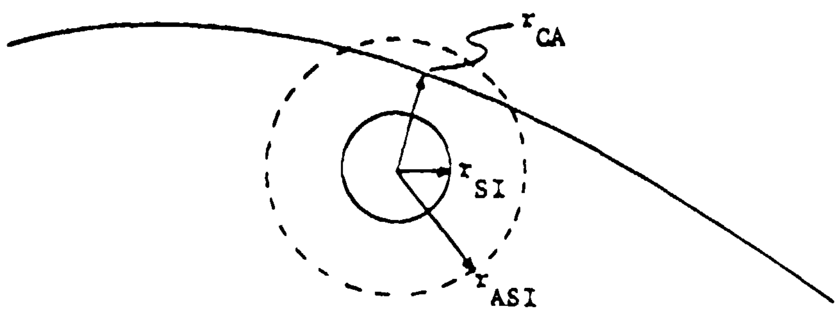
$$(\vec{r}, \vec{v}) = (\vec{r}, \vec{v} + \Delta\vec{v})$$

and the return is made to the iterate loop.

VIII. Outer Targeting

Occasionally the zero iterate initial state leads to a trajectory missing the target body SOI. Since all target options except one (targeting to a specified position, i.e., KTAR = 10,11,12) require the trajectory to intersect the target body SOI steps must be taken to correct this.

Let the initial state propagated forward lead to a trajectory with a closest approach to the target body of r_{CA} with $r_{CA} > r_{SI}$ where r_{SI} is the radius of the SOI.



Until the initial trajectory intersects the SOI the usual targeting can not be done. Therefore an "artificial" SOI is introduced having a radius of

$$r_{ASI} = 1.2 \times r_{CA}$$

The initial trajectory obviously intersects the artificial SOI and hence may be targeted to conditions on the ASOI. If the target conditions are established as $B \cdot T_A = B \cdot R_A = 0$, when this artificial targeting is completed, the refined trajectory will be headed straight for the target body when it hits the ASOI. Thus the refined trajectory should automatically hit the normal SOI when propagated past the ASOI. To insure that the time of intersection with the normal SOI is consistent with the target time, an artificial target time is also used. Let the speed of the spacecraft with respect to the target body at r_{CA} be v_{CA}. Make the approximation that this speed will be roughly the same for the refined trajectory. Then the time that the spacecraft should intersect the ASOI is

$$t_{ASI} = t_{CA} - \frac{r_{ASI}}{v_{CA}}$$

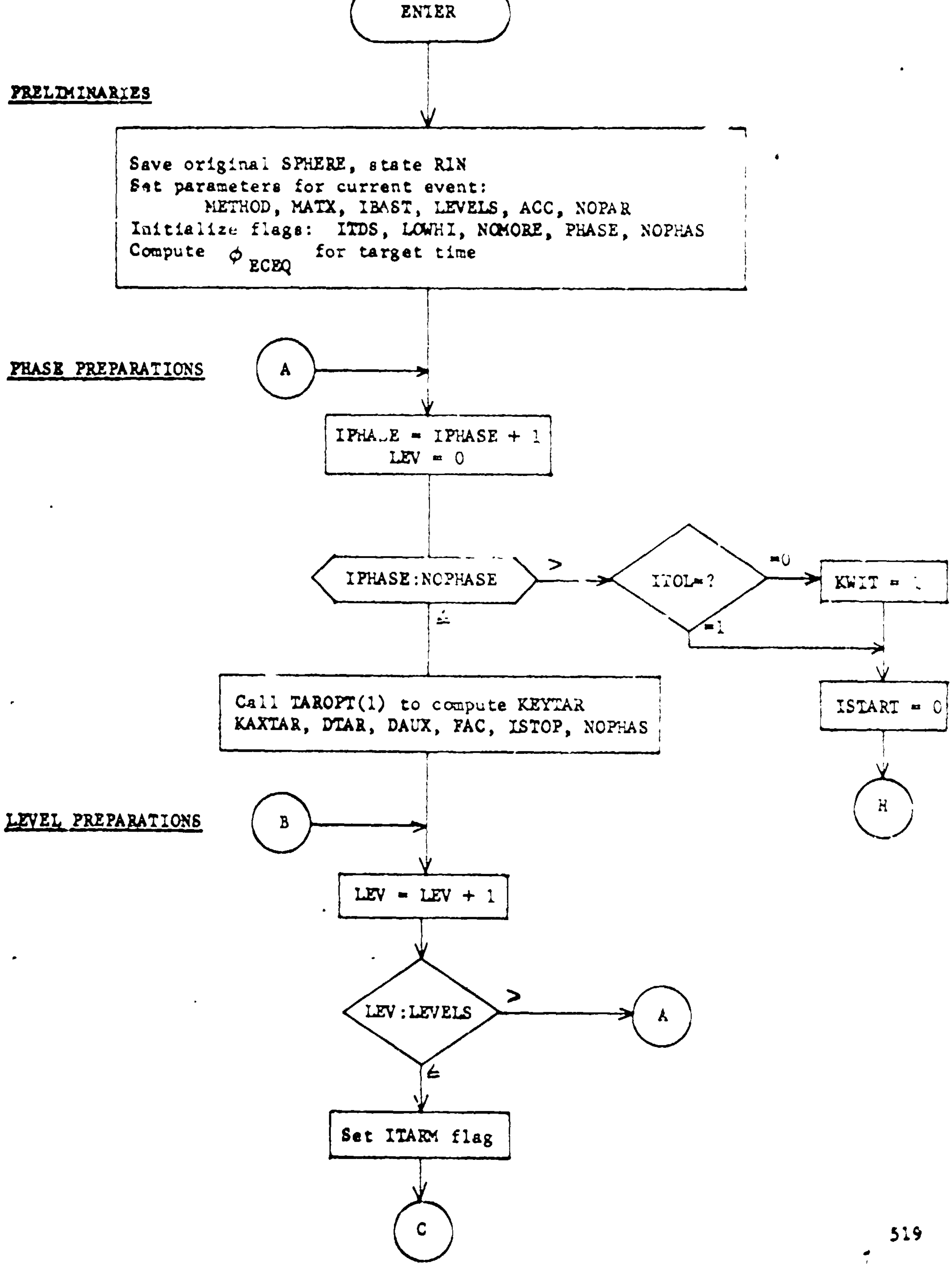
or

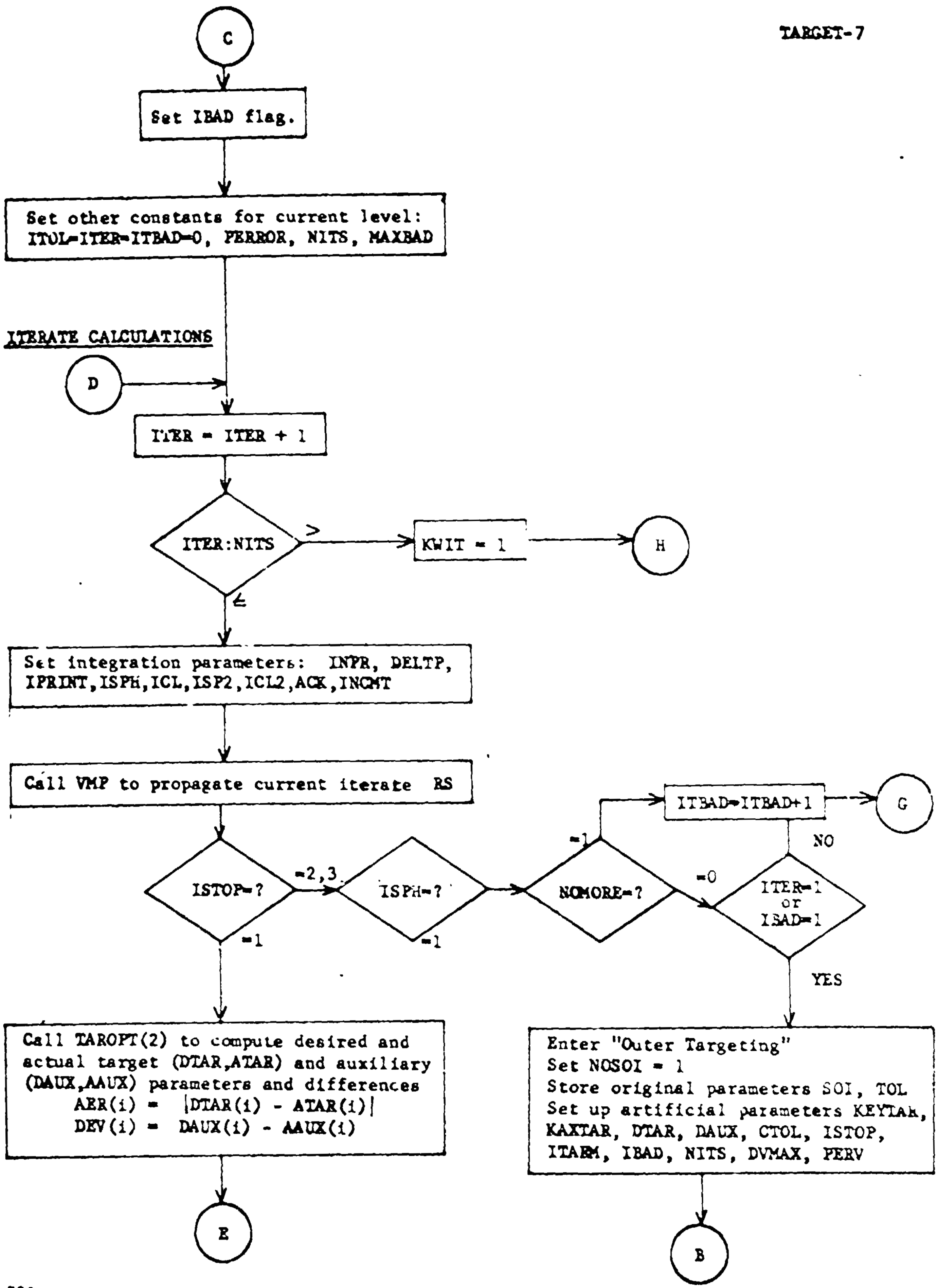
$$t_{ASI} = t_{SI} - \frac{r_{ASI} - r_{SI}}{v_{CA}}$$

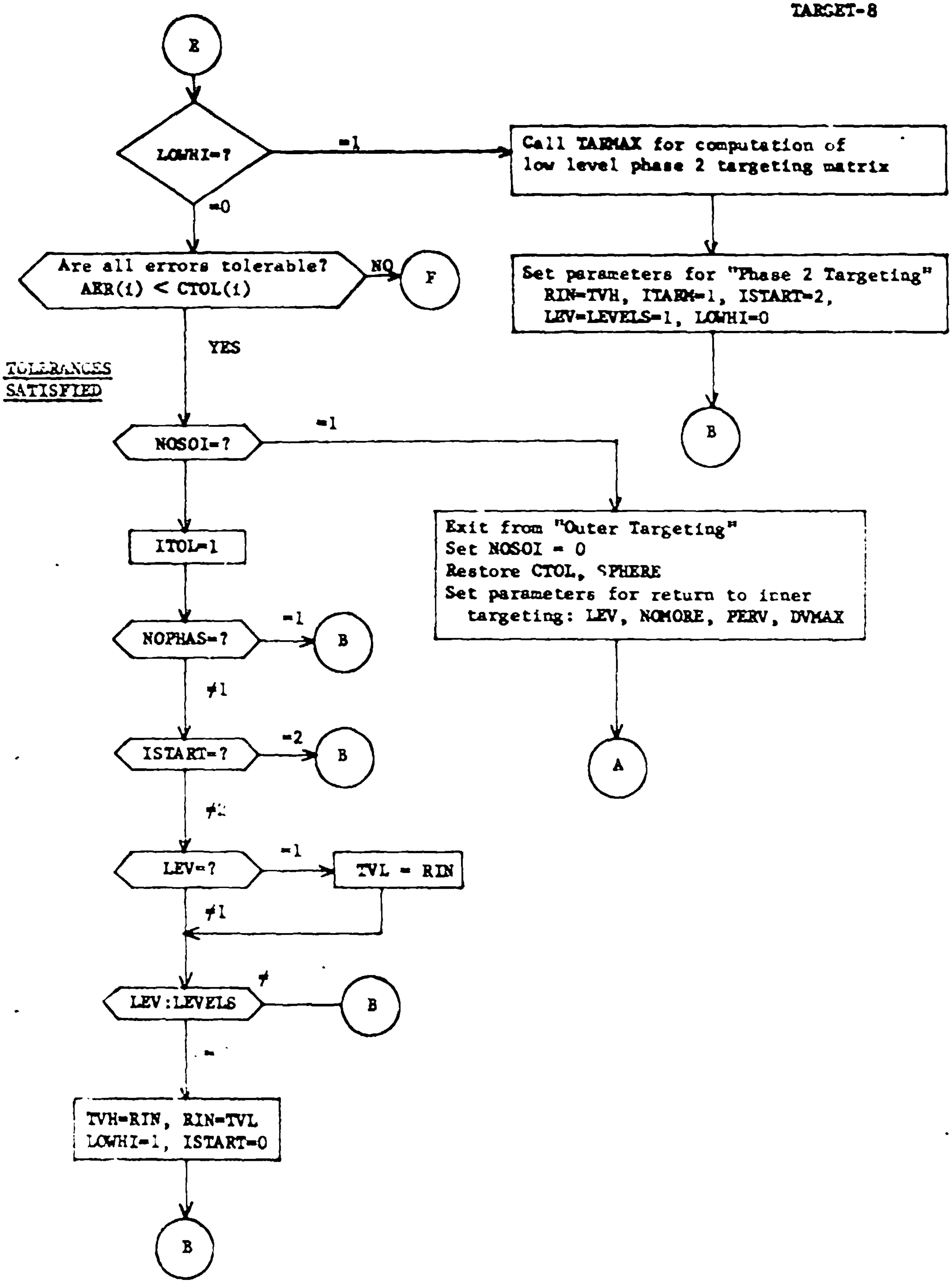
where the first formula should be used if the target time is t_{CA} or t_{CS} and the second formula is used for t_{SI}.

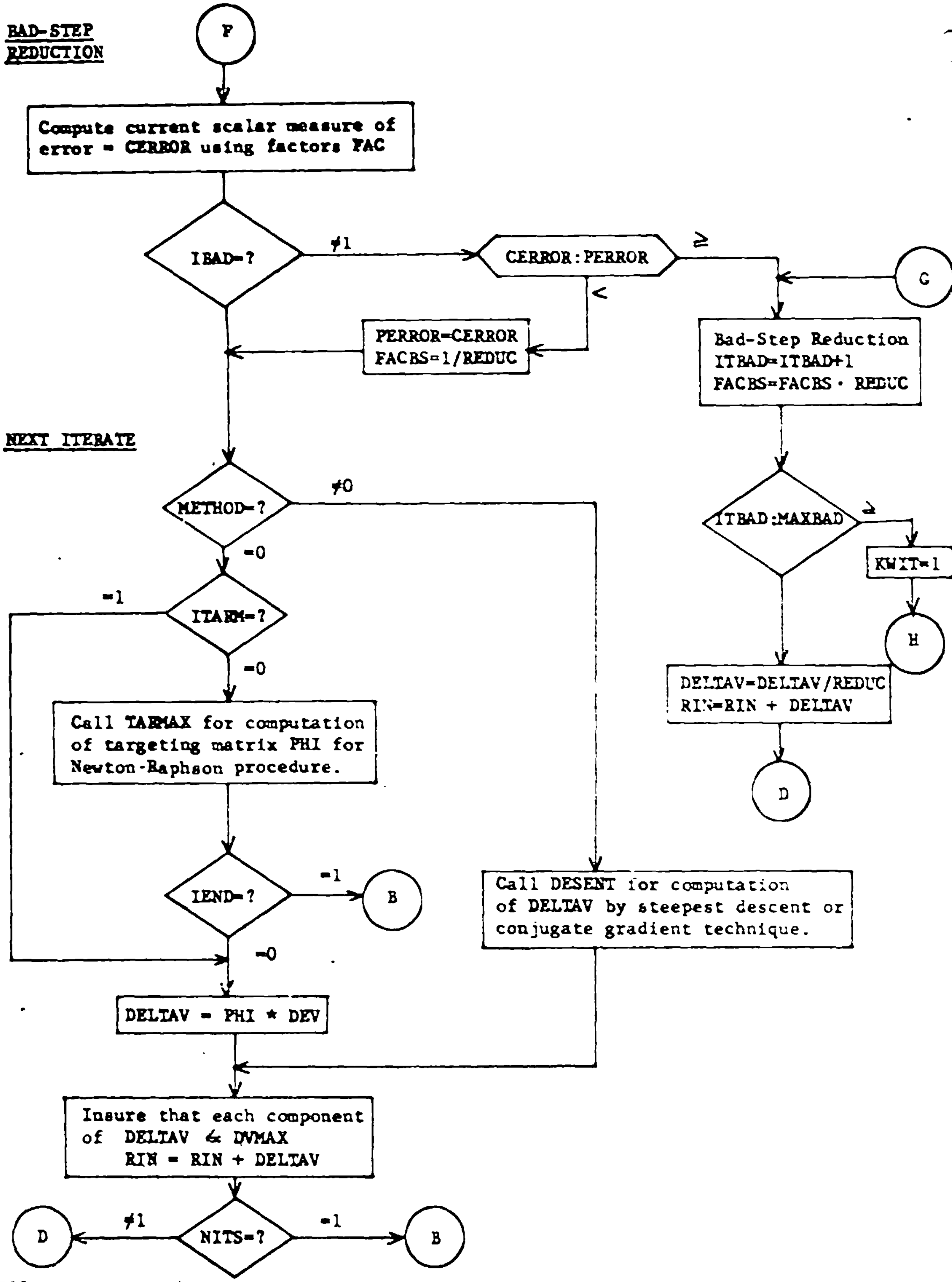
Thus when a trajectory is found which misses the normal SOI, the closest approach state r_{CA}, v_{CA} is recorded. The normal SOI radius is stored and the artificial SOI radius given above is used in its place. Target parameters of $B \cdot T_A$, $B \cdot R_A$, and t_{ASI} are then set up as the targets. When targeting of this artificial problem is complete, the resulting trajectory will intersect the normal SOI and the original problem may be solved.

TARGET Flow Chart

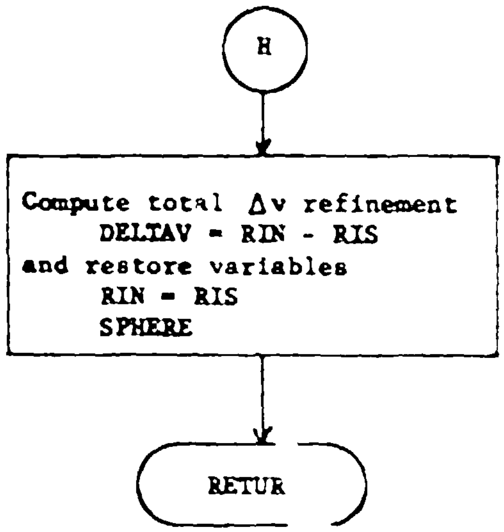



C
Set IBAD flag.
Set other constants for current level:
ITOL=ITER=ITBAD=0, PERROR, NITS, MAXBAD
ITERATE CALCULATIONS
D
ITER = ITER + 1
ITER:NITS
>
KWIT = 1
H
≤
Set integration parameters: INPR, DELTP,
IPRINT, ISPH, ICL, ISP2, ICL2, ACK, INCMT
Call VMP to propagate current iterate RS
ISTOP=?
=2,3
ISPH=?
=1
NOMORE=?
=1
ITBAD=ITBAD+1
G
NO
=0
ITER=1
or
IBAD=1
YES
=1
Call TAROPT(2) to compute desired and
actual target (DTAR,ATAR) and auxiliary
(DAUX,AAUX) parameters and differences
AER(i) = |DTAR(i) - ATAR(i)|
DEV(i) = DAUX(i) - AAUX(i)
Enter "Outer Targeting"
Set NOSOI = 1
Store original parameters SOI, TOL
Set up artificial parameters KEYTAR,
KAXTAR, DTAR, DAUX, CTOL, ISTOP,
ITARM, IBAD, NITS, DVMAX, PERV
E
B


B
LOWHI=?
=1
Call TARMAX for computation of low level phase 2 targeting matrix
=0
Set parameters for "Phase 2 Targeting"
RIN=TVH, ITARM=1, ISTART=2,
LEV=LEVELS=1, LOWHI=0
B
Are all errors tolerable?
AER(i) < CTOL(i)
NO
F
YES
TOLERANCES SATISFIED
NOSOI=?
=1
Exit from "Outer Targeting"
Set NOSOI = 0
Restore CTOL, SPHERE
Set parameters for return to inner targeting: LEV, NOMORE, PERV, DVMAX
A
ITOL=1
NOPHAS=?
=1
B
≠1
ISTART=?
=2
B
≠2
LEV=?
=1
TVL = RIN
≠1
LEV:LEVELS
≠
B
=
TVH=RIN, RIN=TVL
LOWHI=1, ISTART=0
B


BAD-STEP REDUCTION
F
Compute current scalar measure of error = CERROR using factors FAC
IBAD=?
≠1
CERROR:PERROR
≥
G
<
PERROR=CERROR
FACBS=1/REDUC
Bad-Step Reduction
ITBAD=ITBAD+1
FACBS=FACBS · REDUC
NEXT ITERATE
METHOD=?
≠0
=0
ITBAD:MAXBAD
≥
KWIT=1
H
=1
ITARM=?
=0
DELTAV=DELTAV/REDUC
RIN=RIN + DELTAV
Call TARMAX for computation of targeting matrix PHI for Newton-Raphson procedure.
D
IEND=?
=1
B
Call DESENT for computation of DELTAV by steepest descent or conjugate gradient technique.
=0
DELTAV = PHI * DEV
Insure that each component of DELTAV ≤ DVMAX
RIN = RIN + DELTAV
D
≠1
NITS=?
=1
B

PREPARATIONS FOR RETURN



SUBROUTINE TARMAX

PURPOSE: TO CALCULATE A TARGET MATRIX FROM NOMINAL INJECTION CONDITIONS, AND A PERTURBATION FACTOR BDELV FOR A GIVEN ACCURACY LEVEL.

CALLING SEQUENCE: CALL TARMAX

SUBROUTINES SUPPORTED: TARGET

SUBROUTINES REQUIRED: MATIN TAROPT VMP

LOCAL SYMBOLS:

ACK	ACCURACY USED TO GENERATE THE TARGET MATRIX
AER	DIFFERENCES BETWEEN DESIRED AND ACTUAL END CONDITIONS
AUXN	NOMINAL AUXILIARY END CONDITIONS
CHI	STATE TRANSITION MATRIX RELATING PERTURBATIONS IN THE RIN VECTOR TO CHANGES IN AUXN
DVEE	VECTOR OF VELOCITY COMPONENT PERTURBATIONS
ISP2	INDICATOR USED BY SUBROUTINE VMP =0 DO NOT STOP AT SHPERE OF INFLUENCE =1 STOP AT SPHERE OF INFLUENCE
I	INDEX
J	INDEX
KOMP	INDEX
PSI	TARGET MATRIX FOR 2 X 2 EASE, STORED INTO PHI
RSF	FINAL SPACECRAFT STATE RETURNED BY VMP

COMMON COMPUTED/USED: ISPH PHI RIN TRTM

COMMON COMPUTED: ICL2 ICL INCMT

COMMON USED:

AAUX	AC	ATAR	CTOL	DAUX
DELTAT	DELTAV	DTAR	D1	ISTOP
KUR	LEV	LVLS	NOPAR	PERV
ZERO				

TARMAX Analysis

TARMAX computes the targeting matrix used by TARGET for Newton-Raphson refinements. The targeting matrix is computed by numerical differencing.

Let the current iterate initial state be denoted $\vec{r}, \vec{v}$. Let the auxiliary parameters corresponding to this state be $\vec{\alpha}$. Let the perturbation size for the sensitivities be Δv.

The k-th column of the sensitivity matrix is computed as follows. Perturb the k-th component of velocity by Δv:

$$\vec{v}_p = \vec{v} + \Delta v \left[\delta_{1K}, \delta_{2K}, \delta_{3K}\right]^T \tag{1}$$

Propagate the perturbed initial state $(\vec{r}, \vec{v}_p)$ to the final stopping conditions. Let the auxiliary parameters of that trajectory be denoted α_p. The k-th column of the sensitivity matrix x is then given by

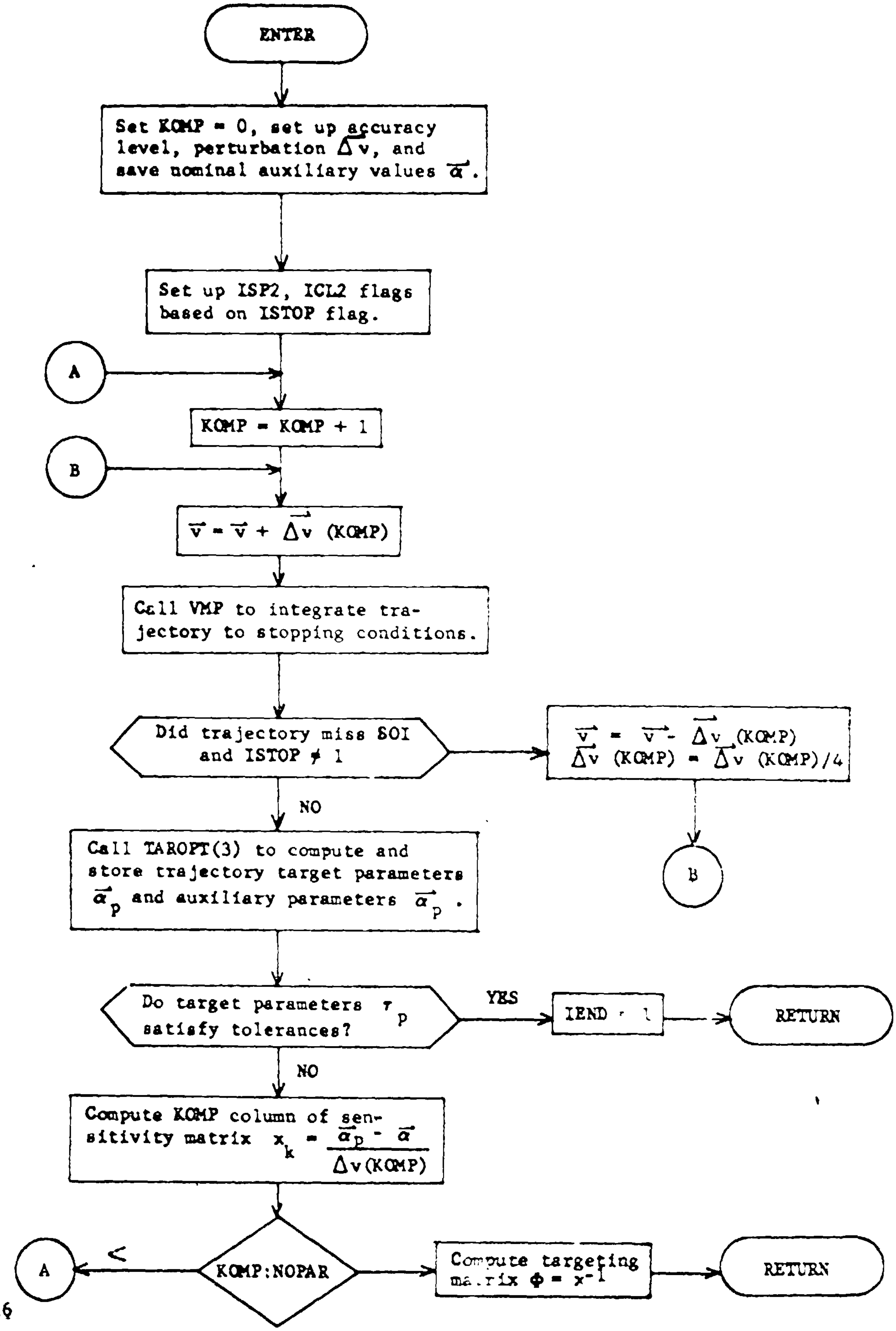
$$x_k = \frac{\vec{\alpha}_p - \vec{\alpha}}{\Delta v} \tag{2}$$

Having computed all the columns of x, the targeting matrix is then given by the inverse of x:

$$\Phi = x^{-1} \tag{3}$$

The targeting matrix then has the property that to obtain a change $\Delta\alpha$ in the nominal auxiliary parameters, the velocity should be changed by the amount

$$\overrightarrow{\Delta v} = \Phi \cdot \overrightarrow{\Delta\alpha} \tag{4}$$


ENTER
Set KOMP = 0, set up accuracy level, perturbation Δv, and save nominal auxiliary values $\vec{\alpha}$.
Set up ISP2, ICL2 flags based on ISTOP flag.
A
KOMP = KOMP + 1
B
$\vec{v} = \vec{v} + \vec{\Delta v}$ (KOMP)
Call VMP to integrate trajectory to stopping conditions.
Did trajectory miss SOI and ISTOP ≠ 1
$\vec{v} = \vec{v} - \vec{\Delta v}$ (KOMP)
$\vec{\Delta v}$ (KOMP) = $\vec{\Delta v}$ (KOMP)/4
B
NO
Call TAROPT(3) to compute and store trajectory target parameters $\vec{\alpha}_p$ and auxiliary parameters $\vec{\alpha}_p$.
Do target parameters τ_p satisfy tolerances?
YES
IEND = 1
RETURN
NO
Compute KOMP column of sensitivity matrix $x_k = \dfrac{\vec{\alpha}_p - \vec{\alpha}}{\Delta v(\text{KOMP})}$
A
<
KOMP:NOPAR
Compute targeting matrix $\Phi = x^{-1}$
RETURN

SUBROUTINE TAROPT

PURPOSE: TO COMPUTE THE DESIRED AND ACHIEVED TARGET PARAMETER VALUES FOR ALL THE TARGETING SUBROUTINES.

CALLING SEQUENCE: CALL TAROPT(ITARO)

ARGUMENT: ITARO I OPTION FLAG
=1 SET UP TARGETING PARAMETERS FOR TARGET KEYS
=2 COMPUTE ACTUAL VALUES OF PARAMETERS
=3 COMPUTE ACTUAL AND DESIRED VALUES OF PARAMETERS

SUBROUTINES SUPPORTED: TARGET TARMAX DESENT

SUBROUTINES REQUIRED: CAREL CPWMS IMPACT

LOCAL SYMBOLS:

ACK	CURRENT ACCURACY BEING USED
A	SEMI-MAJOR AXIS OF THE TARGET PLANETOCENTRIC CONIC
CPT	TOTAL COMPUTER TIME USED (SECS)
DBOR	DESIRED VALUE OF B DOT R
DBOT	DESRIED VALUE OF B DOT T
DINC	DESIRED VALUE OF INCLINATION
DRCA	DESIRED VALUE OF RCA
E	ECCENTRICITY OF THE TARGET PLANETOCENTRIC CONIC
IAUX	INDICATOR FOR AUXILIARY END CONDITIONS =0 TARGET TO ACTUAL END CONDITIONS =1 TARGET TO AUXILIARY END CONDITIONS
IINC	LOCATES DESIRED INCLINATION IN THE DTAR ARRAY
IRCA	LOCATES DESIRED RCA IN THE DTAR ARRAY
I	INDEX
KEY	LOCAL VARIABLE USED TO COMPLETE INFORMATION IN THE KAXTAR AND KEYTAR ARRAY
PP	DUMMY VARIABLE FOR CALL TO CAREL
QQ	DUMMY VARIABLE FOR CALL TO CAREL

RM	MAGNITUDE OF SPACECRAFT USED TO COMPUTE SEMI-MAJOR AXIS
TA	DUMMY ARGUMENT FOR CALL TO CAREL
TDBR	DUMMY ARGUEMENT FOR CALL TO IMPACT
TBDT	DUMMY ARGUEMENT FOR CALL TO IMPACT
TFP	TIME OF FLIGHT FROM PERIAPSIS ON THE TARGET PLANETOCENTRIC CONIC
TIMG	INTERMEDIATE VARIABLE TO COMPUTE CPT
TSICA	DUMMY VARIABLE FOR CALL TO IMPACT
VX	INTERMEDIATE VARIABLE USED TO CALCULATE SEMI-MAJOR AXIS FOR OPTION 9
WW	DUMMY VARIABLE FOR CALL TO CAREL
W	DUMMY VARIABLE FOR CALL TO CAREL
XI	DUMMY VARIABLE FOR CALL TO CAREL
XN	DUMMY VARIABLE FOR CALL TO CAREL

COMMON COMPUTED/USED:	AAUX	ATAR	DAUX	DELTAT	DTAR
	ISTOP	KAXTAR	KEYTAR	NOPAR	NOPHAS
	RCA				
COMMON COMPUTED:	CTOL	FAC			
COMMON USED:	AC	BDR	BDT	CAINC	DC
	DG	DSI	DT	EQECP	IBAD
	ICL2	INCMT	IPHASE	KTAR	KUR
	LEV	NOSOI	NPAR	ONE	RC
	RIN	RRF	RSI	TAR	TIMS
	TMU	TM	TOL	TWO	VSI

TARØPT Analysis

TARØPT is responsible for computing the desired and achieved target parameter values for all the targeting subroutines. Thus to add any new target parameters TARØPT is the only subroutine that must be modified.

The key variables used by TARØPT and their definitions are

Variable	Definition
KTAR(6,10)	Codes of target parameters of all targeting events
TAR(6,10)	Desired values of target parameters of all targeting events
KEYTAR(3)	Codes of target parameters of current event
DTAR(3)	Desired values of target parameters of current event
ATAR(3)	Actual values of target parameters of current iterate
KAXTAR(3)	Codes of auxiliary parameters of current iterate
DAUX(3)	Desired values of auxiliary parameters on current iterate
AAUX(3)	Actual values of auxiliary parameters of current iterate

The available target parameters and their codes and definitions are

Code	Parameter	Definition
1		Available for use
2	t_{SI}	Time at SOI of target body (n-body integration to SOI)
3	t_{CS}	Time at CA (n-body integration to SOI, conic propagation to CA)
4	t_{CA}	Time at CA (n-body integration to CA)
5	$B \cdot T$	Impact parameter $B \cdot T$
6	$B \cdot R$	Impact parameter $B \cdot R$
7	i	Inclination to target planet equator
8	r_{CA}	Radius of closest approach to target body
9	a_{SI}	Semi-major axis of conic w.r.t target body
10	x_f	X-component of final state (inertial ecliptic system)
11	y_f	Y-component of final state
12	z_f	Z-component of final state

The term target parameter refers to a variable whose final value is to conform to a desired value. The term auxiliary parameter refers to a variable which is used to compute the progressive corrections. The target parameters and auxiliary parameters are identical unless the target parameters i and r_{CA} are used. In this case the more linear variables $B \cdot T$ and $B \cdot R$ are used in their place as auxiliary parameters. The desired values of $B \cdot T$ and $B \cdot R$ are then computed (by IMPACT) based on the desired values of i and r_{CA} and the approach asymptote.

TAROPT is called under three different options distinguished by an argument ITARO. The three different options will be discussed in order.

TAROPT(1) is called by TARGET at the beginning of each phase to set up the proper variables for the targeting. The arrays KEYTAR, KAXTAR, DTAR, and DAUX are set to the current event values of KTAR and TAR. If t_{CA} is a target parameter, the number of phases NOPHAS is set to 2. If the current phase is the first phase of a two-phase problem, t_{CA} is replaced by t_{CS} in the KEYTAR and KAXTAR arrays. If i and r_{CA} are target parameters, the corresponding indices of the KAXTAR array are set up for $B \cdot T$ and $B \cdot R$. TAROPT then sets up the integration parameters. The integration time interval Δt is set to the nominal difference of the current guidance event time and target time:

$$\Delta t = t_T - t_G$$

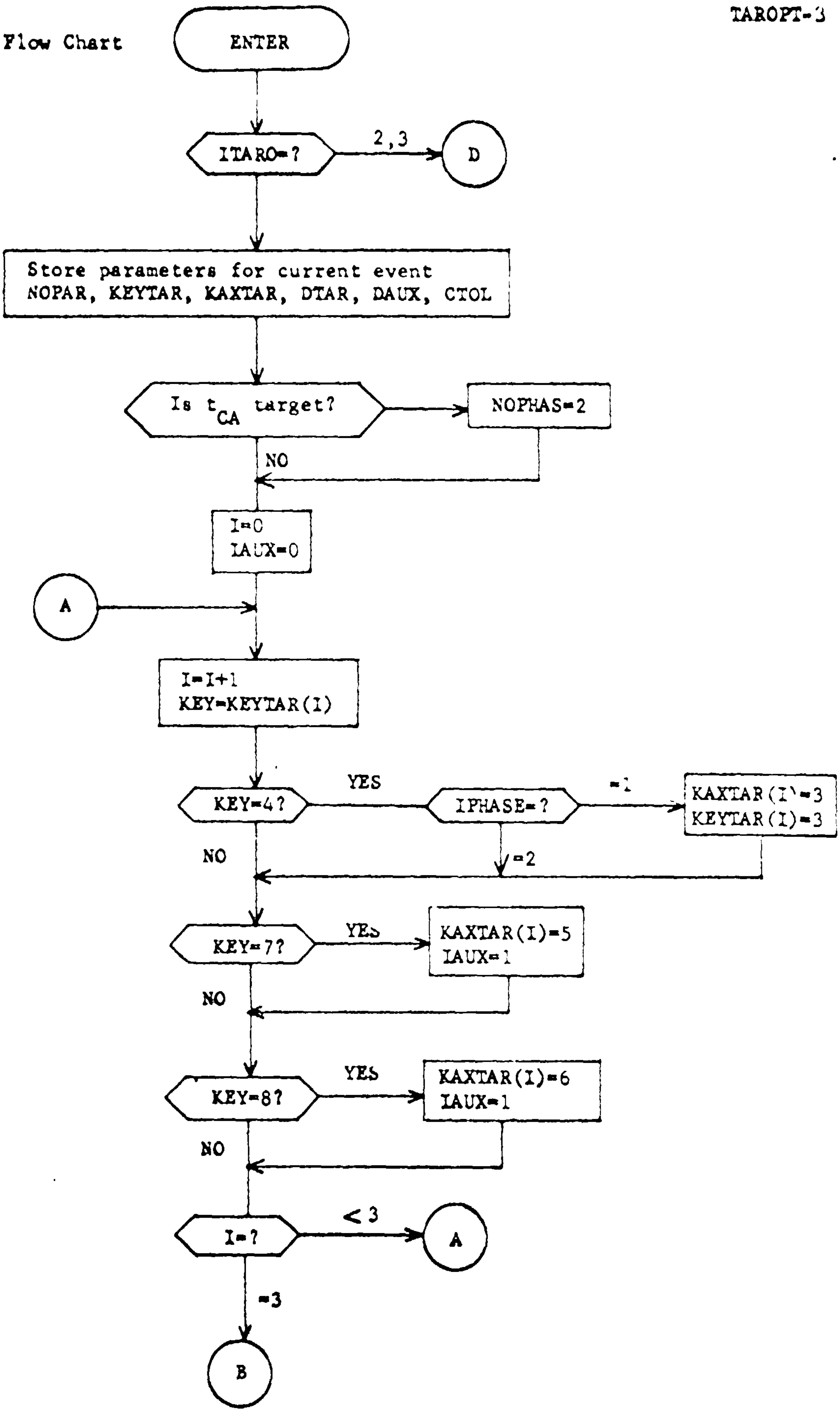
Then if none of the target times are triggered ISTOP is set to 1 so that the integration proceeds to the target time exactly. If the target time is t_{SI} or t_{CS}, ISTOP is set to 2 and $\Delta t = 1.1 \Delta t$. Thus the integration will be stopped at the target body SOI. Finally if the target time is t_{CA}, ISTOP is set to 3 and $\Delta t = 1.1 \Delta t$. For this case the integration will be stopped at closest approach to the target body. Finally the weighting factors FAC(3) to be used in computing the scalar loss function are set. Since all auxiliary parameters are units of length except for the time parameters only the relative weight of time to length need be input. Thus the length factors are set to unity, the time factor is set to the input parameter WGHTM.

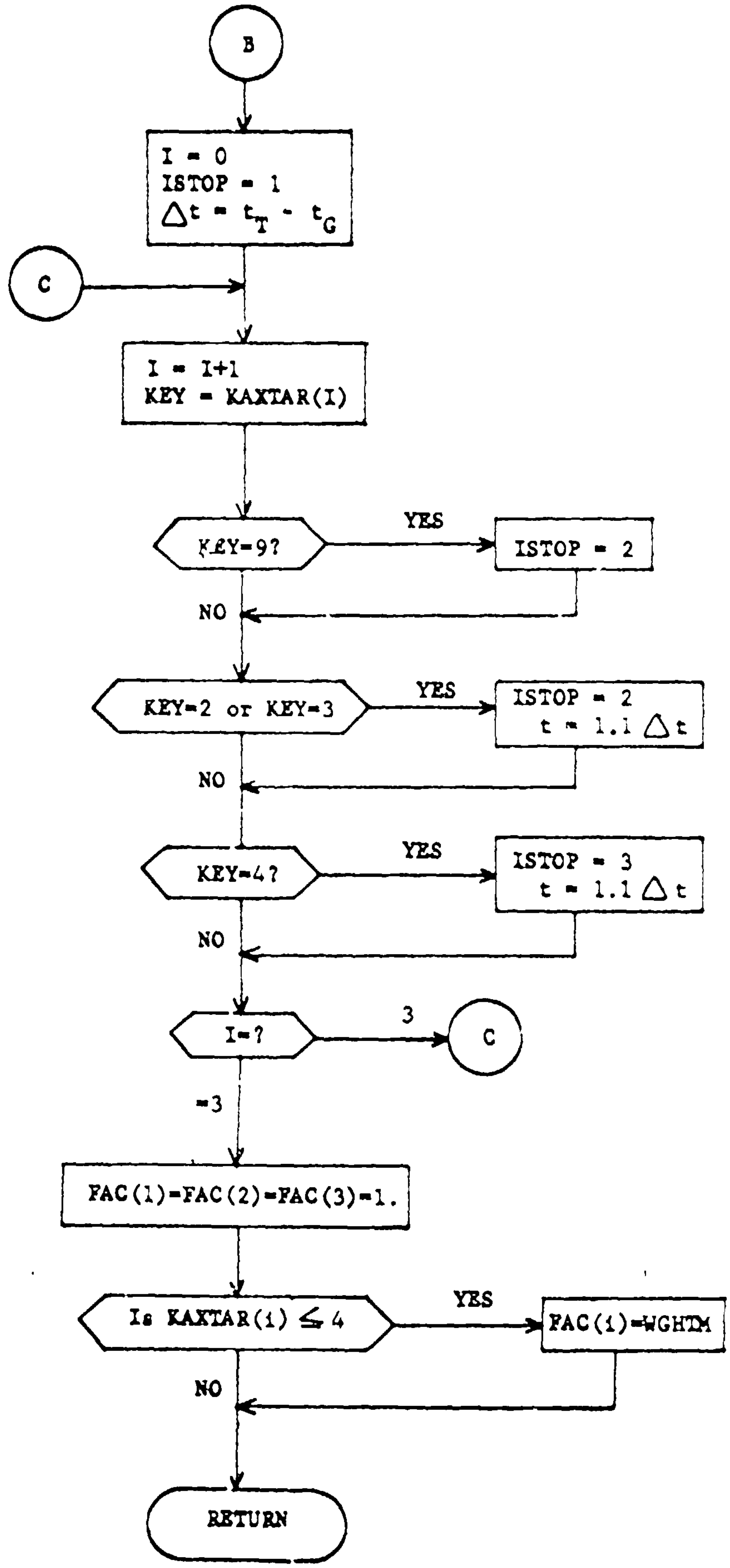
TAROPT(2) is called by TARGET after integrating each iterate to the final stopping conditions. Here TAROPT perform mainly a bookkeeping role. It must fill the proper cells of the ATAR, AAUX, and DAUX arrays with values generally computed by the virtual mass routines. The desired values of $B \cdot T^*$ and $B \cdot R^*$ are computed by calling IMPACT if needed.

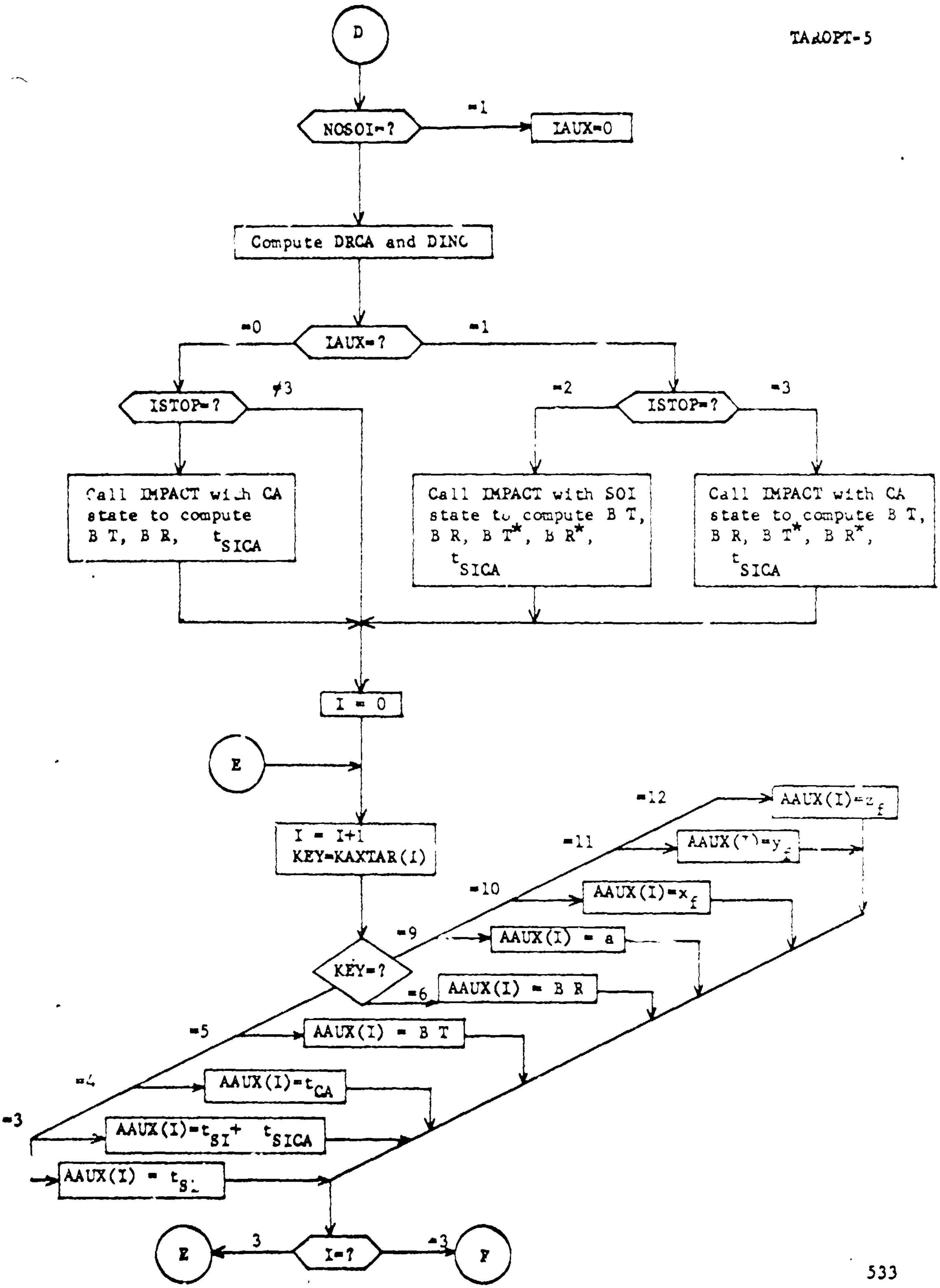
TAROPT(3) is called by TARMAX and DESENT after integrating each perturbed trajectory to compute the perturbed values of the auxiliary parameters. Thus the desired values of DAUX need not be computed at this time. Once again, this task is simply a bookkeeping job to store the trajectory data correctly in the ATAR and AAUX cells. TARMAX and DESENT may then operate easily on these arrays to compute the targeting matrix or gradient directions.

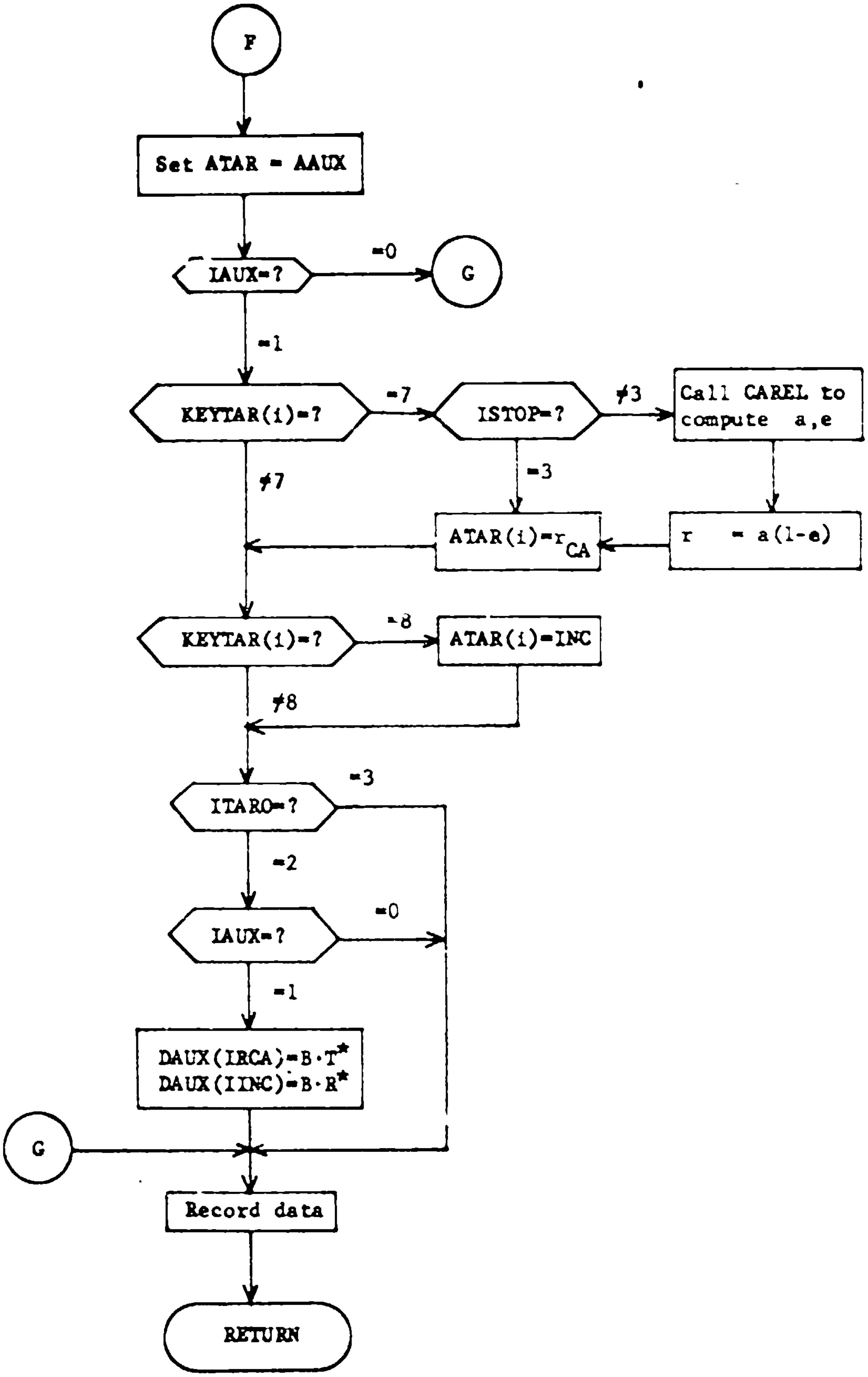
In both calls TAROPT(2) and TAROPT(3) the trajectory data are printed out before exiting from TAROPT.

TAROPT Flow Chart




B
I = 0
ISTOP = 1
Δt = t_T - t_G
C
I = I+1
KEY = KAXTAR(I)
KEY=9?
YES
ISTOP = 2
NO
KEY=2 or KEY=3
YES
ISTOP = 2
t = 1.1 Δt
NO
KEY=4?
YES
ISTOP = 3
t = 1.1 Δt
NO
I=?
3
C
=3
FAC(1)=FAC(2)=FAC(3)=1.
Is KAXTAR(i) ≤ 4
YES
FAC(i)=WGHTM
NO
RETURN


D
NOSOI=?
=1
IAUX=0
Compute DRCA and DINC
=0
IAUX=?
=1
ISTOP=?
≠3
=2
ISTOP=?
=3
Call IMPACT with CA state to compute B T, B R, tSICA
Call IMPACT with SOI state to compute B T, B R, B T*, B R*, tSICA
Call IMPACT with CA state to compute B T, B R, B T*, B R*, tSICA
I = 0
E
I = I+1
KEY=KAXTAR(I)
KEY=?
=12
AAUX(I)=zf
=11
AAUX(I)=yf
=10
AAUX(I)=xf
=9
AAUX(I) = a
=6
AAUX(I) = B R
=5
AAUX(I) = B T
=4
AAUX(I)=tCA
=3
AAUX(I)=tSI+ tSICA
AAUX(I) = tSI
E
3
I=?
=3
F


F
Set ATAR = AAUX
IAUX=?
=0
G
=1
KEYTAR(i)=?
=7
ISTOP=?
≠3
Call CAREL to compute a,e
=3
≠7
ATAR(i)=r_{CA}
r = a(1-e)
KEYTAR(i)=?
=8
ATAR(i)=INC
≠8
ITARO=?
=3
=2
IAUX=?
=0
=1
DAUX(IRCA)=B·T*
DAUX(IINC)=B·R*
G
Record data
RETURN

SUBROUTINE TARPRL

PURPOSE: TO COMPUTE THE PARTIAL DERIVATIVES OF THE POSITION COMPONENTS OF A PLANET WITH RESPECT TO EACH OF ITS ORBITAL ELEMENTS.

CALLING SEQUENCE: CALL TARPRL(ICODE,PAR)

ARGUMENT: ICODE I CODE DEFINING ORBITAL ELEMENT OF INTEREST

PAR O VECTOR OF 3 POSITION PARTIALS WITH RESPECT TO THE ORBITAL ELEMENT OF INTEREST

SUBROUTINES SUPPORTED: TRAKS TRAKM

LOCAL SYMBOLS:

CBO	COSINE OF LONGITUDE OF ASCENDING NODE
CI	COSINE OF ANGLE OF INCLINATION
CLO	COSINE OF ARGUMENT OF PERIAPSIS
COSNU	COSINE OF TRUE ANOMALY
COSONU	COSINE OF THE SUM OF THE ANGLES OF TRUE ANOMALY PLUS THE ARGUMENT OF PERIAPSIS
DNUDE	PLANET DISTANCE TIMES THE PARTIAL OF TRUE ANOMALY WITH RESPECT TO ECCENTRICITY
DNUDM	PARTIAL OF TRUE ANOMALY WITH RESPECT TO MEAN ANOMALY
DPAR	1./R*(≠R/≠E + (≠R/≠NU)*(≠NU/≠E)) WHERE R= PLANET DISTANCE NU=TRUE ANOMALY E= ECCENTRICITY ≠= PARTIAL OF
DRDNU	PARTIAL R WITH RESPECT TO NU
E2	SQUARE OF ECCENTRICITY
IND	INDEX USED IN ARRAY STORING ORBITAL ELEMENTS OF PLANETS
PCOMP	SEMI-MAJOR AXIS TIMES THE TERM (1-E*E) WHERE E=ECCENTRICITY
R	PLANET DISTANCE
SBO	SINE OF LONGITUDE OF THE ASCENDING NODE
SI	SINE OF INCLINATION

SINNU	SINE OF TRUE ANOMALY
SINONU	SINE OF THE SUM OF THE ANGLES OF TRUE ANOMALY PLUS THE ARGUMENT OF PERIAPSIS
SLO	SINE OF THE ARGUMENT OF PERIAPSIS
XXX	SCRATCH CELL

COMMON USED: ALNGTH ELMNT NTP ONE XP ZERO

TARPRL Analysis

The position components of a planet are related to its orbital elements a, e, i, Ω, ω, and M through the following set of equations:

$$x = r\left[\cos\Omega\cos(\omega+\nu) - \sin\Omega\sin(\omega+\nu)\cos i\right] \quad (1)$$

$$y = r\left[\sin\Omega\cos(\omega+\nu) + \cos\Omega\sin(\omega+\nu)\cos i\right] \quad (2)$$

$$z = r\sin(\omega+\nu)\sin i \quad (3)$$

$$r = \frac{a(1-e^2)}{1+e\cos\nu} \quad (4)$$

$$\tan\frac{\nu}{2} = \sqrt{\frac{1+e}{1-e}}\tan\frac{E}{2} \quad (5)$$

$$M = E - e\sin E \quad (6)$$

We can write equations (1), (2), (3), and (4) symbolically as

$$x_i = f_i(a, e, i, \Omega, \omega, \nu)$$

and equations (5) and (6) as

$$\nu = \nu(e, M) \quad .$$

Then the partials of x_i with respect to a, e, i, Ω, ω, and M can be evaluated as follows:

$$\frac{\partial x_i}{\partial a} = \frac{\partial f_i}{\partial a} \quad (7)$$

$$\frac{\partial x_i}{\partial e} = \left(\frac{\partial f_i}{\partial e}\right)_\nu + \frac{\partial f_i}{\partial \nu}\cdot\frac{\partial \nu}{\partial e} \quad (8)$$

$$\frac{\partial x_i}{\partial i} = \frac{\partial f_i}{\partial i} \quad (9)$$

$$\frac{\partial x_i}{\partial \Omega} = \frac{\partial f_i}{\partial \Omega} \quad (10)$$

$$\frac{\partial x_i}{\partial \omega} = \frac{\partial f_i}{\partial \omega} \tag{11}$$

$$\frac{\partial x_i}{\partial M} = \frac{\partial f_i}{\partial \nu} \cdot \frac{\partial \nu}{\partial M} \tag{12}$$

Only $\frac{\partial \nu}{\partial e}$ and $\frac{\partial \nu}{\partial M}$ require further consideration before equations (7) through (11) can be used to obtain expressions for the 18 desired partial derivatives.

We obtain $\frac{\partial \nu}{\partial M}$ by first differentiating equation (5) with respect to E and equation (6) with respect to M to obtain

$$\frac{\partial \nu}{\partial E} = \frac{a}{r} \sqrt{1 - e^2}$$

and

$$\frac{\partial E}{\partial M} = \frac{a}{r} \quad .$$

Then

$$\frac{\partial \nu}{\partial M} = \frac{\partial \nu}{\partial E} \cdot \frac{\partial E}{\partial M} = \left(\frac{a}{r}\right)^2 \sqrt{1 - e^2} \quad . \tag{13}$$

We obtain $\frac{\partial \nu}{\partial e}$ by first differentiating equation (6) with respect to e to obtain

$$\frac{\partial M}{\partial e} = - \frac{\sqrt{1 - e^2} \sin \nu}{1 + e \cos \nu} \quad .$$

This result is then combined with equation (13) to yield

$$\frac{\partial \nu}{\partial e} = \frac{\partial \nu}{\partial M} \frac{\partial M}{\partial e} = - \left(\frac{a}{r}\right)^2 \frac{(1 - e^2) \sin \nu}{1 + e \cos \nu} \tag{14}$$

The evaluation of the desired partials can now proceed. The results are summarized below.

a. Partials with respect to a.

$$\frac{\partial x}{\partial a} = \frac{x}{a}$$

$$\frac{\partial y}{\partial a} = \frac{y}{a}$$

$$\frac{\partial z}{\partial a} = \frac{z}{a}$$

b. Partials with respect to e.

$$\frac{\partial x}{\partial e} = \frac{xq}{r} + r\frac{\partial \nu}{\partial e}\left[-\cos\Omega\sin(\omega+\nu) - \sin\Omega\cos(\omega+\nu)\cos i\right]$$

$$\frac{\partial y}{\partial e} = \frac{yq}{r} + r\frac{\partial \nu}{\partial e}\left[-\sin\Omega\sin(\omega+\nu) + \cos\Omega\cos(\omega+\nu)\cos i\right]$$

$$\frac{\partial z}{\partial e} = \frac{zq}{r} + r\frac{\partial \nu}{\partial e}\cos(\omega+\nu)\sin i$$

$$\text{where} \quad q = \frac{r}{ae(1-e^2)}\left[r - a - ae^2(1+\sin^2\nu)\right]$$

c. Partials with respect to i.

$$\frac{\partial x}{\partial i} = r\sin\Omega\sin(\omega+\nu)\sin i$$

$$\frac{\partial y}{\partial i} = -r\cos\Omega\sin(\omega+\nu)\sin i$$

$$\frac{\partial z}{\partial i} = r\sin(\omega+\nu)\cos i$$

d. Partials with respect to Ω.

$$\frac{\partial x}{\partial \Omega} = -y$$

$$\frac{\partial y}{\partial \Omega} = x$$

$$\frac{\partial z}{\partial \Omega} = 0$$

e. Partials with respect to ω.

$$\frac{\partial x}{\partial \omega} = r\left[-\cos\Omega\sin(\omega+\nu) - \sin\Omega\cos(\omega+\nu)\cos i\right]$$

$$\frac{\partial y}{\partial \omega} = r\left[-\sin\Omega\sin(\omega+\nu) + \cos\Omega\cos(\omega+\nu)\cos i\right]$$

$$\frac{\partial z}{\partial \omega} = r\cos(\omega+\nu)\sin i$$

f. Partials with respect to M.

$$\frac{\partial x}{\partial M} = \frac{xs}{r} + r\frac{\partial \nu}{\partial M}\left[-\cos\Omega\sin(\omega+\nu) - \sin\Omega\cos(\omega+\nu)\cos i\right]$$

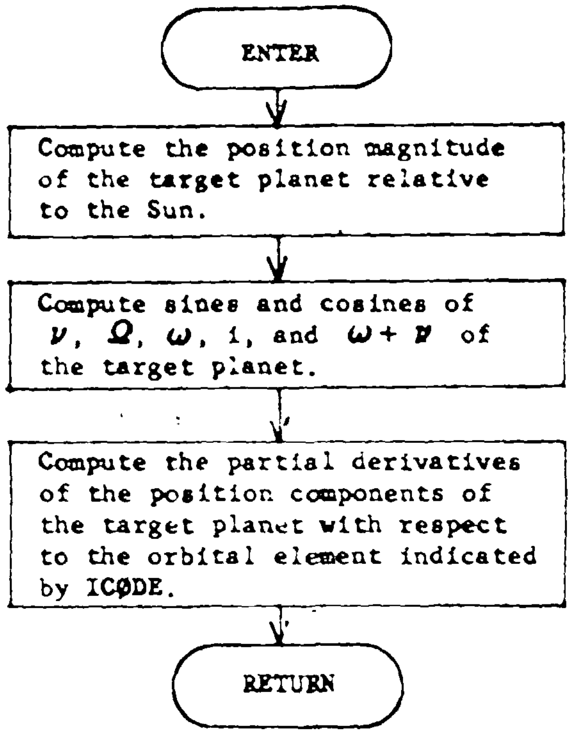
$$\frac{\partial y}{\partial M} = \frac{ys}{r} + r\frac{\partial \nu}{\partial M}\left[-\sin\Omega\sin(\omega+\nu) + \cos\Omega\cos(\omega+\nu)\cos i\right]$$

$$\frac{\partial z}{\partial M} = \frac{zs}{r} + r\frac{\partial \nu}{\partial M}\cos(\omega+\nu)\sin i$$

where $s = \dfrac{ae\sin\nu}{\sqrt{1-e^2}}$

Reference: Battin, R. H.: Astronautical Guidance, McGraw-Hill Book Company, Inc., New York, 1964.

TARPRL Flow Chart



SUBROUTINE TIME

PURPOSE: TO COMPUTE THE JULIAN DATE, EPOCH 1900, FROM THE CALENDAR DATE OR TO COMPUTE THE CALENDAR DATE FROM THE JULIAN DATE.

CALLING SEQUENCE: CALL TIME(DAY,IYR,MO,IDAY,IHR,MIN,SEC,ICODE)

ARGUMENT: DAY I/O JULIAN DATE, EPOCH 1900

IYR O/I CALENDAR YEAR

MO O/I CALENDAR MONTH

IDAY O/I CALENDAR DAY

IHR O/I HOUR OF THE DAY

MIN O/I MINUTE OF HOUR

SEC O/I FRACTIONAL SECONDS

ICODE I OPERATIONAL MODE
= 1, INDICATES THE JULIAN DATE IS INPUT, CALENDAR DATE IS OUTPUT
= 0, INDICATES THE CALENDAR DATE IS INPUT, JULIAN DATE IS OUTPUT

SUBROUTINES SUPPORTED: DATAS INPUTZ PRINT VMP GIDANS PREPUL PRNTS4 DATA PRNTS3 PRELIM GIDANS HELIO MULTAR

SUBROUTINES REQUIRED: NONE

LOCAL SYMBOLS: IA NUMBER OF CENTURIES

IB YEARS IN PRESENT CENTURY

IP NUMBER OF MONTH (BASED ON MARCH AS NUMBER ZERO)

IQ NUMBER OF YEARS

IR NUMBER OF CENTURIES DIVIDED BY 4

IS NUMBER OF YEARS SINCE LAST 400 YEAR SECTION BEGAN

IT NUMBER OF LEAP YEARS IN PRESENT CENTURY

IU NUMBER OF YEARS SINCE LAST LEAP YEAR

IV NUMBER OF DAYS IN LAST YEAR

IX	INTERMEDIATE INTEGER
J	INTERMEDIATE INTEGER
JD	NUMBER OF DAYS IN JULIAN DATE
P	JULIAN DATE
R	FRACTIONAL PORTION OF DAY IN JULIAN DATE

SUBROUTINE TITLE

PURPOSE: TO PRINT TITLES FOR ERRAN.

CALLING SEQUENCE: CALL TITLE(LINES,TEVN,ICODE)

ARGUMENT:	LINES		NOT USED
	TEVN	I	EVENT TIME
	ICODE	I	EVENT CODE

SUBROUTINES SUPPORTED: SETEVN

LOCAL SYMBOLS: TPT TIME PREDICTING TO

COMMON USED: IPROB NPE TPT2

```
SUBROUTINE   TITLES

PURPOSE:  TO PRINT TITLES FOR SIMUL.

CALLING SEQUENCE:  CALL TITLES(TEVN,ICODE)

ARGUMENT:    TEVN     I   EVENT TIME

             ICODE    I   EVENT CODE

SUBROUTINES SUPPORTED: SETEVS

LOCAL SYMBOLS:  TPT      TIME PREDICTING TO


COMMON USED:             IPROB   NPE     TPT2
```

SUBROUTINE TRAKM

PURPOSE: THE OBSERVATIONS AND OBSERVATION MATRIX FOR A GIVEN TYPE OF MEASUREMENT IS COMPUTED BY THIS ROUTINE

CALLING SEQUENCE: CALL TRAKM(HECV,ITRK,NR,IOBS,VECTOR)

ARGUMENT:

HECV	I	POSITION AND VELOCITY OF SPACECRAFT AT TIME OF MEASUREMENT
IOBS	I	CODE WHICH SPECIFIES IF MEASUREMENT OR OBSERVATION MATRIX IS TO BE COMPUTED
ITRK	I	CODE WHICH SPECIFIES MEASUREMENT TYPE (CALLED MMCODE ELSEWHERE IN PROGRAM)
NR	O	NUMBER OF ROWS IN THE OBSERVATION MATRIX
VECTOR	O	ACTUAL MEASUREMENT

SUBROUTINES SUPPORTED: ERRANN

SUBROUTINES REQUIRED: EPHEM ORB STAPRL TARPRL

LOCAL SYMBOLS:

AD1	INTERMEDIATE VARIABLE
AD2	INTERMEDIATE VARIABLE
AD3	INTERMEDIATE VARIABLE
AL	ALTITUDE
ALAT	LATITUDE
ALON	LONGITUDE
A1	PARTIAL OF RANGE WITH RESPECT TO X
A2	PARTIAL OF RANGE WITH RESPECT TO Y
A3	PARTIAL OF RANGE WITH RESPECT TO Z
B1	PARTIAL OF RANGE-RATE WITH RESPECT TO X
B2	PARTIAL OF RANGE-RATE WITH RESPECT TO Y
B3	PARTIAL OF RANGE-RATE WITH RESPECT TO Z
CE	COSINE OF OBLIQUITY OF EARTH
COAL	COSINE OF STAR-PLANET ANGLE
CP	COSINE OF LONGITUDE + CONSTANT

DADP	PARTIALS OF STAR-PLANET ANGLE WITH RESPECT TO VEHICLE POSITION AND VELOCITY
DBDP	PARTIALS OF APPARENT PLANET DIAMETER WITH RESPECT TO VEHICLE POSITION AND VELOCITY
DENOM	INTERMEDIATE VARIABLE
D	INTERMEDIATE TIME
EK	RANGE-RATE PARTIAL WITH RESPECT TO STATION LOCATION ERRORS
GECS	GEOCENTRIC EQUATORIAL COORDINATES OF STATION
GELS	GEOCENTRIC ECLIPTIC COORDINATES [illegible]
HECE	COORDINATES OF EARTH
HECP	COORDINATES OF TARGET PLANET
IA	TRACKING STATION LOCATION SELECTION CODE
ICD	CODE CORRESPONDING TO TRACKING [illegible] LOCATION ERRORS
IC	COLUMN NUMBER IN OBSERVATION MATRIX PARTITION WHERE EK IS TO BE STORED
IEND	VARIABLE INDEX VALUE
IR	STAR-PLANET ANGLE INDEX INCREMENT VALUE
NA	STAR-PLANET ANGLE INDEX LOWER [illegible] =1 FOR 3 STAR-PLANET ANGLES =ITRK-10 FOR SINGLE STAR-PLANET [illegible]
NC	STAR-PLANET ANGLE INDEX UPPER LIMIT =3 FOR 3 STAR-PLANET ANGLES =ITRK-10 FOR SINGLE STAR-PLA[illegible]
PAR	PARTIALS RETURNED FROM SUBROUTINE [illegible]
PAT1	INTERMEDIATE VARIABLE
PAT2	INTERMEDIATE VARIABLE
PA	PARTIALS
RADNTP	RADIUS OF TARGET PLANET

RRATE	RANGE-RATE
R1	RANGE
R2	SQUARE OF RANGE
SA	PARTIALS OF STAR-PLANET ANGLES WITH RESPECT TO VEHICLE POSITION
SE	SINE OF OBLIQUITY OF EARTH
SIAL	SINE OF STAR-PLANET ANGLE
SP	SINE OF LONGITUDE + CONSTANT
VEC	INTERMEDIATE VECTOR

COMMON COMPUTED/USED:

AAL	AM	H	NO	T
XP				

COMMON COMPUTED: G

COMMON USED:

ALNGTH	DATEJ	DELTM	EM3	EPS
F	IAUGDC	IAUGIN	IAUGMC	IAUG
IBARY	NBOD	NB	NTP	OMEGA
ONE	RADIUS	SAL	SLAT	SLON
TM	TRTM1	TWO	UNIVT	UST
VST	WST	ZERO		

TRAKM Analysis

Subroutine TRAKM computes observation matrix partitions in the error analysis mode. It is completely equivalent to the simulation mode subroutine TRAKS with IØBS always set to zero. See subroutine TRAKS for further analytical details. A flow chart is not presented for TRAKM since it is but a subset of the TRAKS flow chart.

SUBROUTINE TRAKS

PURPOSE: TO COMPUTE ALL OBSERVATION MATRIX PARTITIONS FOR THE MEASUREMENT TYPE AND TO COMPUTE THE MEASUREMENT ITSELF.

CALLING SEQUENCE: CALL TRAKS(HECV,ITRK,NR,IOBS,VECTOR)

ARGUMENT:	HECV	I	POSITION AND VELOCITY OF SPACECRAFT AT TIME OF MEASUREMENT
	IOBS	I	CODE WHICH SPECIFIES IF MEASUREMENT OR OBSERVATION MATRIX IS TO BE COMPUTED
	ITRK	I	CODE WHICH SPECIFIES MEASUREMENT TYPE (CALLED MMCODE ELSEWHERE IN PROGRAM)
	NR	O	NUMBER OF ROWS IN THE OBSERVATION MATRIX
	VECTOR	O	ACTUAL MEASUREMENT

SUBROUTINES SUPPORTED: SIMULL

SUBROUTINES REQUIRED: EPHEM ORB STAPRL TARPRL

LOCAL SYMBOLS:	AD1	INTERMEDIATE VARIABLE
	AD2	INTERMEDIATE VARIABLE
	AD3	INTERMEDIATE VARIABLE
	ALAT	LATITUDE
	ALON	LONGITUDE
	AL	ALTITUDE
	A1	PARTIAL OF RANGE WITH RESPECT TO X
	A2	PARTIAL OF RANGE WITH RESPECT TO Y
	A3	PARTIAL OF RANGE WITH RESPECT TO Z
	B1	PARTIAL OF RANGE-RATE WITH RESPECT TO X
	B2	PARTIAL OF RANGE-RATE WITH RESPECT TO Y
	B3	PARTIAL OF RANGE-RATE WITH RESPECT TO Z
	CE	COSINE OF OBLIQUITY OF EARTH
	COAL	COSINE OF STAR-PLANET ANGLE
	CP	COSINE OF LONGITUDE + CONSTANT

DADP	PARTIALS OF STAR-PLANET ANGLE WITH RESPECT TO VEHICLE POSITION AND VELOCITY
DBDP	PARTIALS OF APPARENT PLANET DIAMETER WITH RESPECT TO VEHICLE POSITION AND VELOCITY
DENOM	INTERMEDIATE VARIABLE
D	INTERMEDIATE TIME
EK	RANGE-RATE PARTIAL WITH RESPECT TO STATION LOCATION ERRORS
GECS	GEOCENTRIC EQUATORIAL COORDINATES OF STATION
GELS	GEOCENTRIC ECLIPTIC COORDINATES OF STATION
HECE	COORDINATES OF EARTH
HECP	COORDINATES OF TARGET PLANET
IA	TRACKING STATION LOCATION SELECTION CODE
ICD	CODE CORRESPONDING TO TRACKING STATION LOCATION ERRORS
IC	COLUMN NUMBER IN OBSERVATION MATRIX PARTITION WHERE IS TO BE STORED
IEND	VARIABLE INDEX VALUE
IR	STAR-PLANET ANGLE INDEX INCREMENT VALUE
NA	STAR-PLANET ANGLE INDEX LOWER LIMIT =1 FOR 3 STAR-PLANET ANGLES =ITRK-10 FOR SINGLE STAR-PLANET ANGLES
NC	STAR-PLANET ANGLE INDEX UPPER LIMIT =3 FOR 3 STAR-PLANET ANGLES =ITRK-10 FOR SINGLE STAR-PLANET ANGLES
PAR	PARTIALS RETURNED FROM SUBROUTINE TARPRL
PAT1	INTERMEDIATE VARIABLE
PAT2	INTERMEDIATE VARIABLE
PA	PARTIALS
RADNTP	RADIUS OF TARGET PLANET

RRATE	RANGE-RATE
R1	RANGE
R2	SQUARE OF RANGE
SA	PARTIALS OF STAR-PLANET ANGLES WITH RESPECT TO VEHICLE POSITION
SE	SINE OF OBLIQUITY OF EARTH
SIAL	SINE OF STAR-PLANET ANGLE
SP	SINE OF LONGITUDE + CONSTANT
VEC	INTERMEDIATE VECTOR

COMMON COMPUTED/USED:	AAL	AM	H	NO	XP
COMMON COMPUTED:	G				
COMMON USED:	ALNGTH	DATEJ	DELTM	EM3	EPS
	F	IAUGDC	IAUGIN	IAUGMC	IAUG
	IBARY	NBOD	NB	NTP	OMEGA
	ONE	RADIUS	SAL	SLAT	SLB
	SLON	TM	TRTMB	TRTM1	TWO
	UNIVT	UST	VST	WST	ZERO

TRAKS Analysis

Subroutine TRAKS performs two functions in the simulation mode. The first function, which corresponds to IØBS = 0, is to compute all observation matrix partitions for the measurement type indicated by ITRK. The second function, which corresponds to IØBS ≠ 0, is to compute the measurement itself. If IØBS = 1, TRAKS computes the measurement corresponding to the most recent nominal spacecraft state. If IØBS = 2, TRAKS computes the measurement corresponding to the actual spacecraft state, and, if the measurement is a range or range-rate measurement, to the actual tracking station locations. The number of rows, NR, in the measurement and the observation matrix partitions is also computed.

A general measurement has form

$$\vec{Y} = \vec{Y}(\vec{X}, \vec{p}, t)$$

where $\vec{X}$ is the spacecraft position/velocity state at time t and $\vec{p}$ is a vector of parameters. This equation can be linearized about nominal $\vec{X}$ and $\vec{p}$ to obtain

$$\delta\vec{Y} = \left(\frac{\partial \vec{Y}}{\partial \vec{X}}\right)^{*} \delta\vec{X} + \left(\frac{\partial \vec{Y}}{\partial \vec{p}}\right)^{*} \delta\vec{p}$$

where $(\ )^{*}$ indicates matrices are evaluated at the nominal condition. This perturbation equation can be rewritten as

$$\delta\vec{Y} = H\,\delta\vec{X} + M\,\delta\vec{x}_s + G\,\delta\vec{u} + L\,\delta\vec{v}$$

where $H = \left(\frac{\partial \vec{Y}}{\partial \vec{X}}\right)^{*}$, and $\left(\frac{\partial \vec{Y}}{\partial \vec{p}}\right)^{*}$ is distributed among the M, G, and L partitions to correspond to the partition of the parameter vector $\delta\vec{p}$ into solve-for parameters $\delta\vec{x}_s$, dynamic consider parameters $\delta\vec{u}$, and measurement consider parameters $\delta\vec{v}$.

In the remainder of this section the measurement equation and all partial derivatives required to construct the H, M, G, and L observation matrix partitions will be summarized for each measurement type.

<u>A. Range measurement ρ</u>.

A range measurement has form

$$\rho = \rho(\vec{X}, R, \theta, \phi, t)$$

where R, θ, and ϕ are the radius, latitude, and longitude of the relevant tracking station.

More explicitly,

$$\rho = \left[(X - X_E - X_S)^2 + (Y - Y_E - Y_S)^2 + (Z - Z_E - Z_S)^2 \right]^{\frac{1}{2}}$$

where X, Y, Z = inertial position components of spacecraft
X_E, Y_E, Z_E = inertial position components of Earth
X_S, Y_S, Z_S = station position components relative to Earth.

X_S, Y_S, and Z_S are related to R, θ, and ϕ as follows:

$$X_S = R \cos \theta \cos G$$

$$Y_S = R \cos \theta \cos \epsilon \sin G + R \sin \theta \sin \epsilon$$

$$Z_S = -R \cos \theta \sin \epsilon \sin G + R \sin \theta \cos \epsilon$$

where ϵ is the obliquity of the Earth, and

$$G = \phi + GHA$$

where GHA is the Greenwich hour angle at time t .

Partials of ρ with respect to spacecraft state are given by

$$\frac{\partial \rho}{\partial X} = \frac{1}{\rho} (X - X_E - X_S) \qquad \frac{\partial \rho}{\partial \dot{X}} = 0$$

$$\frac{\partial \rho}{\partial Y} = \frac{1}{\rho} (Y - Y_E - Y_S) \qquad \frac{\partial \rho}{\partial \dot{Y}} = 0$$

$$\frac{\partial \rho}{\partial Z} = \frac{1}{\rho} (Z - Z_E - Z_S) \qquad \frac{\partial \rho}{\partial \dot{Z}} = 0$$

Partials of ρ with respect to R, θ, and ϕ are given by

$$\frac{\partial \rho}{\partial R} = \frac{\partial \rho}{\partial X_S} \cdot \frac{\partial X_S}{\partial R} + \frac{\partial \rho}{\partial Y_S} \cdot \frac{\partial Y_S}{\partial R} + \frac{\partial \rho}{\partial Z_S} \cdot \frac{\partial Z_S}{\partial R}$$

$$\frac{\partial \rho}{\partial \theta} = \frac{\partial \rho}{\partial X_S} \cdot \frac{\partial X_S}{\partial \theta} + \frac{\partial \rho}{\partial Y_S} \cdot \frac{\partial Y_S}{\partial \theta} + \frac{\partial \rho}{\partial Z_S} \cdot \frac{\partial Z_S}{\partial \theta}$$

$$\frac{\partial \rho}{\partial \phi} = \frac{\partial \rho}{\partial X_S} \cdot \frac{\partial X_S}{\partial \phi} + \frac{\partial \rho}{\partial Y_S} \cdot \frac{\partial Y_S}{\partial \phi} + \frac{\partial \rho}{\partial Z_S} \cdot \frac{\partial Z_S}{\partial \phi}$$

where

$$\frac{\partial \rho}{\partial X_S} = -\frac{\partial \rho}{\partial X}, \quad \frac{\sigma \rho}{\partial Y_S} = -\frac{\partial \rho}{\partial Y}, \quad \frac{\partial \rho}{\partial Z_S} = -\frac{\partial \rho}{\partial Z}$$

and the negatives of the partials of X_S, Y_S, and Z_S with respect to R, Θ, and Ø are summarized in the subroutine STAPRL analysis.

B. Range-rate measurement $\dot{\rho}$.

A range-rate measurement has form

$$\dot{\rho} = \dot{\rho}(\vec{X}, R, \theta, \emptyset, t)$$

where all arguments have been defined previously. More explicitly,

$$\dot{\rho} = \frac{\rho_x \dot{\rho}_x + \rho_y \dot{\rho}_y + \rho_z \dot{\rho}_z}{\rho}$$

where

$$\rho_x = X - X_E - X_S \qquad \dot{\rho}_x = \dot{X} - \dot{X}_E - \dot{X}_S$$

$$\rho_y = Y - Y_E - Y_S \qquad \dot{\rho}_y = \dot{Y} - \dot{Y}_E - \dot{Y}_S$$

$$\rho_z = Z - Z_E - Z_S \qquad \dot{\rho}_z = \dot{Z} - \dot{Z}_E - \dot{Z}_S$$

$\dot{X}_S$, $\dot{Y}_S$, and $\dot{Z}_S$ are related to R, Θ, and Ø as follows:

$$\dot{X}_S = -\omega R \cos\theta \sin G$$

$$\dot{Y}_S = \omega R \cos\theta \cos\epsilon \cos G$$

$$\dot{Z}_S = -\omega R \cos\theta \sin\epsilon \cos G$$

where ω is the rotational rate of the Earth.

Partials of $\dot{\rho}$ with respect to spacecraft state are given by

$$\frac{\partial \dot{\rho}}{\partial X} = \frac{\dot{\rho}_x}{\rho} - \frac{\rho_x \dot{\rho}}{\rho^2} \qquad \frac{\partial \dot{\rho}}{\partial \dot{X}} = \frac{\rho_x}{\rho}$$

$$\frac{\partial \dot{\rho}}{\partial Y} = \frac{\dot{\rho}_y}{\rho} - \frac{\rho_y \dot{\rho}}{\rho^2} \qquad \frac{\partial \dot{\rho}}{\partial \dot{Y}} = \frac{\rho_y}{\rho}$$

$$\frac{\partial \dot{\rho}}{\partial Z} = \frac{\dot{\rho}_z}{\rho} - \frac{\rho_z \dot{\rho}}{\rho^2} \qquad \frac{\partial \dot{\rho}}{\partial \dot{Z}} = \frac{\rho_z}{\rho}$$

The partial of $\dot{\rho}$ with respect to R is given by

$$\frac{\partial\dot{\rho}}{\partial R} = \frac{\partial\dot{\rho}}{\partial X_S}\cdot\frac{\partial X_S}{\partial R} + \frac{\partial\dot{\rho}}{\partial Y_S}\cdot\frac{\partial Y_S}{\partial R} + \frac{\partial\dot{\rho}}{\partial Z_S}\cdot\frac{\partial Z_S}{\partial R} +$$

$$\frac{\partial\dot{\rho}}{\partial \dot{X}_S}\cdot\frac{\partial \dot{X}_S}{\partial R} + \frac{\partial\dot{\rho}}{\partial \dot{Y}_S}\cdot\frac{\partial \dot{Y}_S}{\partial R} + \frac{\partial\dot{\rho}}{\partial \dot{Z}_S}\cdot\frac{\partial \dot{Z}_S}{\partial R}$$

where

$$\frac{\partial\dot{\rho}}{\partial X_S} = -\frac{\partial\dot{\rho}}{\partial X}\text{, etc.}$$

and $$\frac{\partial\dot{\rho}}{\partial \dot{X}_S} = -\frac{\partial\dot{\rho}}{\partial \dot{X}}\text{, etc.}$$

The negatives of the partials of X_S, Y_S, Z_S, $\dot{X}_S$, $\dot{Y}_S$, and $\dot{Z}_S$ with respect to R, θ, and $\emptyset$ are summarized in the subroutine STAPRL analysis. Partials of $\dot{\rho}$ with respect to θ and $\emptyset$ are treated similarly.

C. Star-planet angle measurement α.

A star-planet angle measurement has form α

$$\alpha = \alpha\,(\vec{X}, a, e, i, \Omega, \omega, M)$$

where a, e, i, Ω, ω, and M are the standard set of target planet orbital elements.

If we define $\vec{\rho} = (\rho_x, \rho_y, \rho_z)$ to be the position of the target planet relative to the spacecraft and (u, v, w) to be the direction cosines of the relevant star, then

$$\alpha = \cos^{-1}\left[\frac{1}{\rho}(u\rho_x + v\rho_y + w\rho_z)\right]$$

where

$$\rho_x = X_p - X,\quad \rho_y = Y_p - Y,\quad \rho_z = Z_p - Z,$$

and (X_p, Y_p, Z_p) represent the position coordinates of the target planet.

Partials of α with respect to spacecraft state are given by

$$\frac{\partial \alpha}{\partial X} = \frac{1}{\sin \alpha}\left(\frac{u}{\rho} - \frac{\rho_x \cos \alpha}{\rho^2}\right) \qquad \frac{\partial \alpha}{\partial \dot{X}} = 0$$

$$\frac{\partial \alpha}{\partial Y} = \frac{1}{\sin \alpha}\left(\frac{v}{\rho} - \frac{\rho_y \cos \alpha}{\rho^2}\right) \qquad \frac{\partial \alpha}{\partial \dot{Y}} = 0$$

$$\frac{\partial \alpha}{\partial Z} = \frac{1}{\sin \alpha}\left(\frac{w}{\rho} - \frac{\rho_z \cos \alpha}{\rho^2}\right) \qquad \frac{\partial \alpha}{\partial \dot{Z}} = 0$$

where

$$\sin \alpha = + \left[1 - \cos^2 \alpha\right]^{\frac{1}{2}} .$$

The partial of α with respect to target planet semi-major axis is given by

$$\frac{\partial \alpha}{\partial a} = \frac{\partial \alpha}{\partial X_p} \cdot \frac{\partial X_p}{\partial a} + \frac{\partial \alpha}{\partial Y_p} \cdot \frac{\partial Y_p}{\partial a} + \frac{\partial \alpha}{\partial Z_p} \cdot \frac{\partial Z_p}{\partial a}$$

where $\dfrac{\partial \alpha}{\partial X_p} = -\dfrac{\partial \alpha}{\partial X}$, $\dfrac{\partial \alpha}{\partial Y_p} = -\dfrac{\partial \alpha}{\partial Y}$, $\dfrac{\partial \alpha}{\partial Z_p} = -\dfrac{\partial \alpha}{\partial Z}$,

and partials of X_p, Y_p, and Z_p with respect to semi-major axis are summarized in the subroutine TARPRL analysis. Partials of α with respect to $\dot{X}_p$, $\dot{Y}_p$, and $\dot{Z}_p$ do not appear in the above expression since they are all zero. Partials of α with respect to the remaining target planet orbital elements are treated similarly.

D. Apparent planet diameter measurement β.

An apparent planet diameter measurement has form

$$\beta = \beta(\vec{X}, a, e, i, \Omega, \omega, M)$$

where all arguments have been defined previously.

Defining $\vec{\rho} = (\rho_x, \rho_y, \rho_z)$ to be the position of the target planet relative to the spacecraft and R_p to be the radius of the target planet, the apparent planet diameter can then be written as

$$\beta = 2 \sin^{-1}\left(\frac{R_p}{\rho}\right)$$

Partials of β with respect to spacecraft state are given by

$$\frac{\partial \beta}{\partial X} = \frac{2 R_p \rho_x}{\rho^2 \left[\rho^2 - R_p^2\right]^{\frac{1}{2}}} \qquad \frac{\partial \beta}{\partial \dot{X}} = 0$$

$$\frac{\partial \beta}{\partial Y} = \frac{2 R_p \rho_y}{\rho^2 \left[\rho^2 - R_p^2\right]^{\frac{1}{2}}} \qquad \frac{\partial \beta}{\partial \dot{Y}} = 0$$

$$\frac{\partial \beta}{\partial Z} = \frac{2 R_p \rho_z}{\rho^2 \left[\rho^2 - R_p^2\right]^{\frac{1}{2}}} \qquad \frac{\partial \beta}{\partial \dot{Z}} = 0$$

The partial of β with respect to target planet semi-major axis is given by

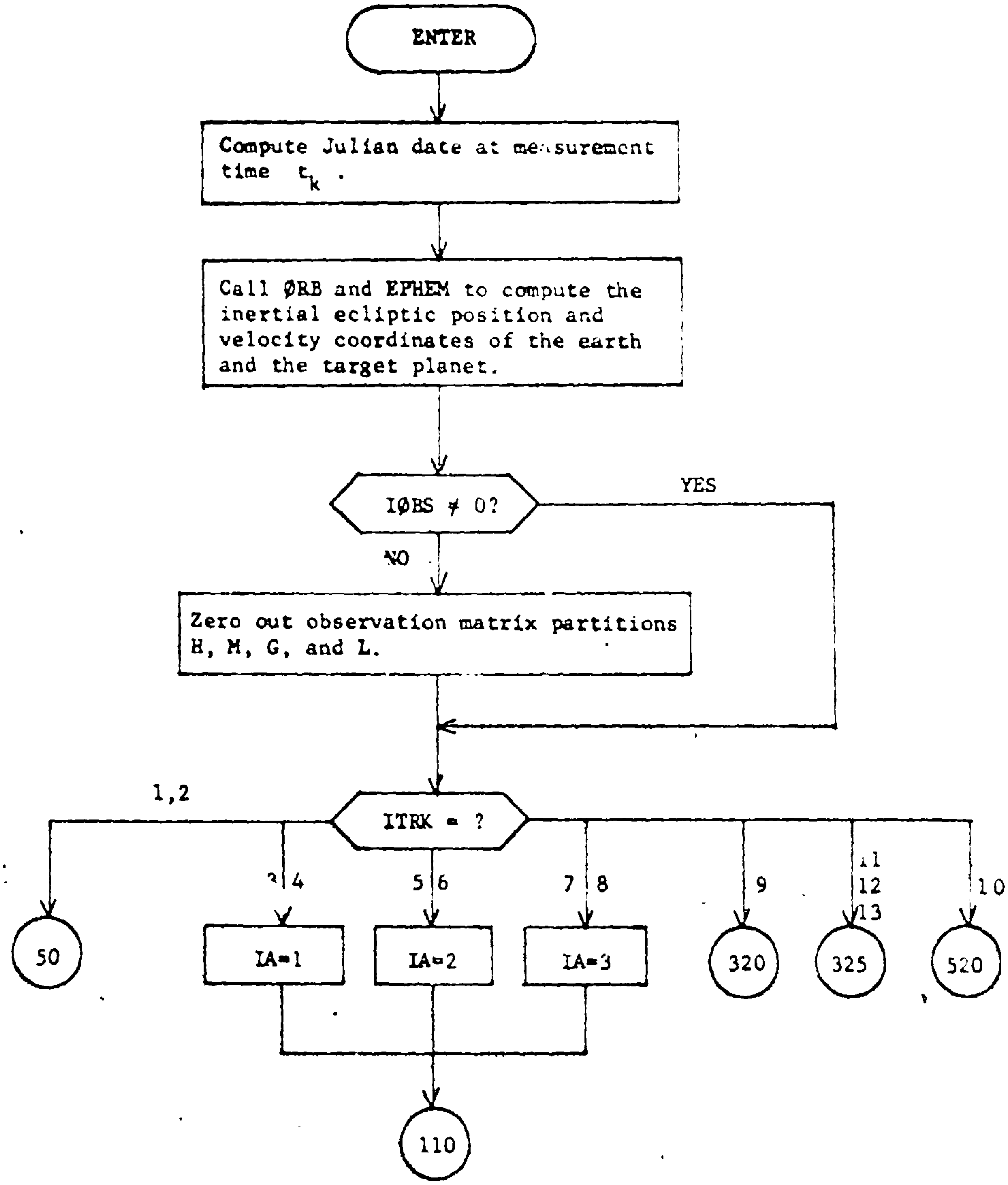
$$\frac{\partial \beta}{\partial a} = \frac{\partial \beta}{\partial X_p} \cdot \frac{\partial X_p}{\partial a} + \frac{\partial \beta}{\partial Y_p} \cdot \frac{\partial Y_p}{\partial a} + \frac{\partial \beta}{\partial Z_p} \cdot \frac{\partial Z_p}{\partial a}$$

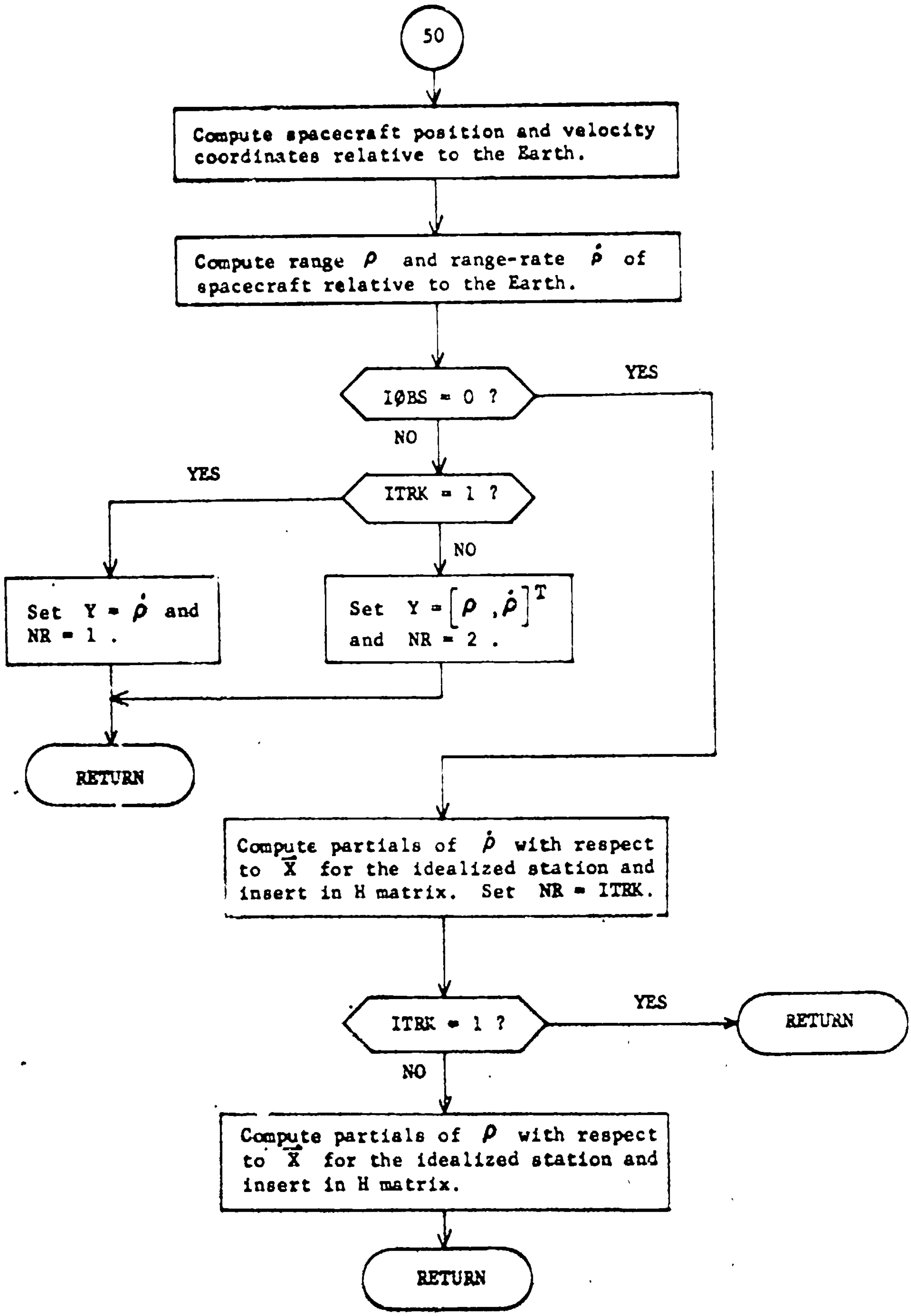
where $\frac{\partial \beta}{\partial X_p} = -\frac{\partial \beta}{\partial X}$, $\frac{\partial \beta}{\partial Y_p} = -\frac{\partial \beta}{\partial Y}$, $\frac{\partial \beta}{\partial Z_p} = -\frac{\partial \beta}{\partial Z}$,

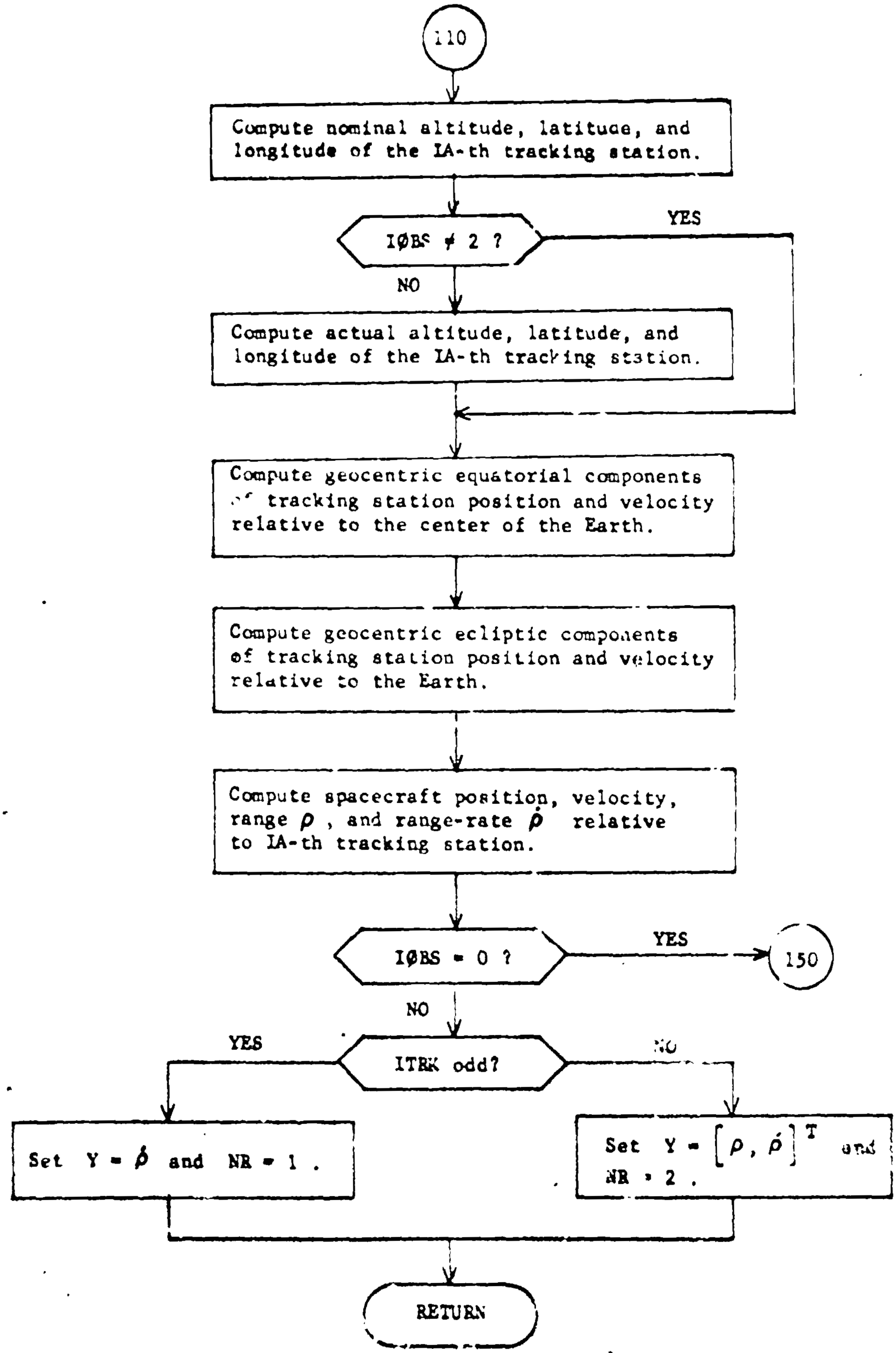
and partials of X_p, Y_p, and Z_p with respect to semi-major axis are summarized in the subroutine TARPRL analysis.

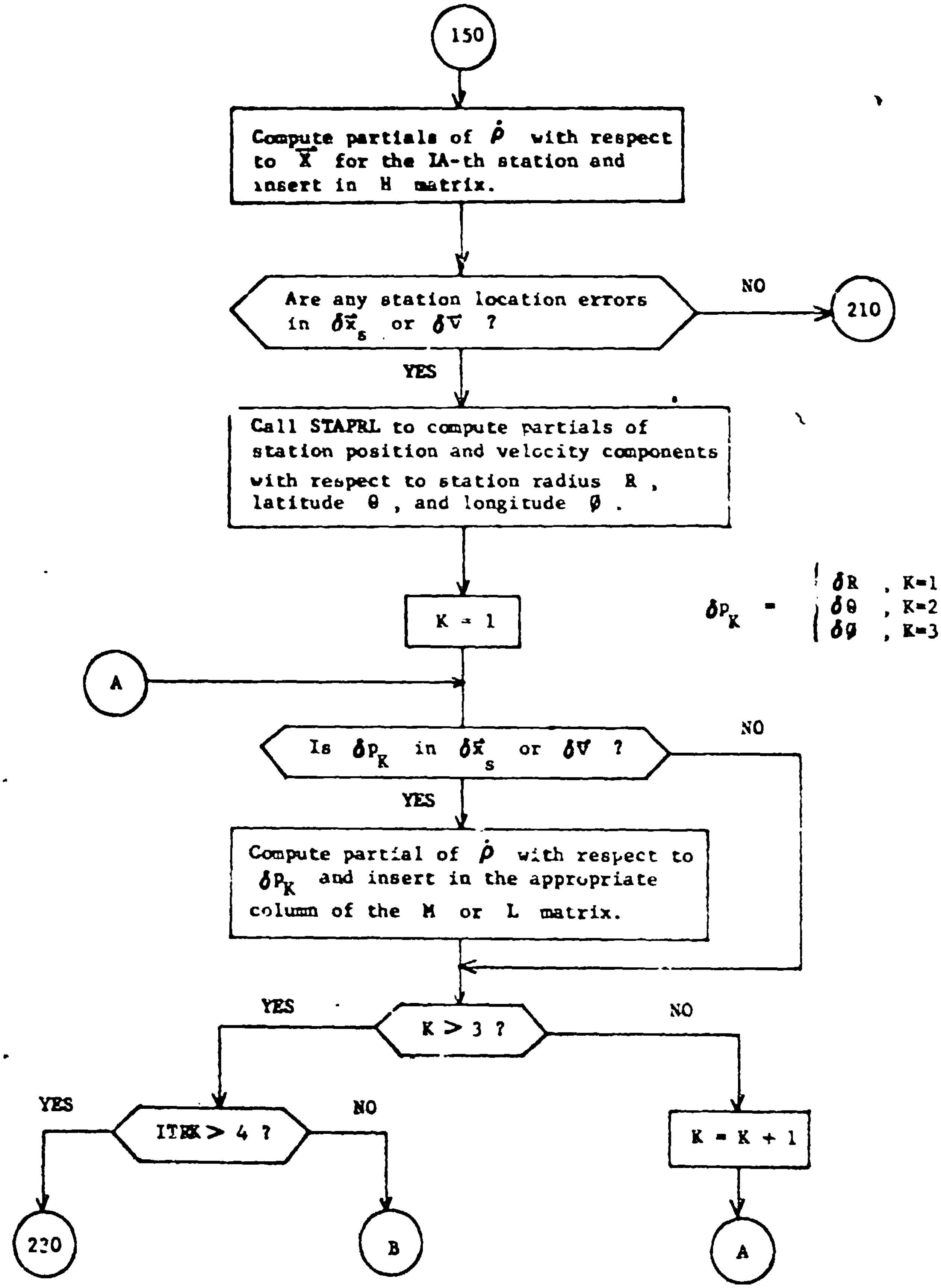
Partials of β with respect to $\dot{X}_p$, $\dot{Y}_p$, and $\dot{Z}_p$ do not appear in the above expression since they are all zero. Partials of β with respect to the remaining target planet orbital elements are treated similarly.

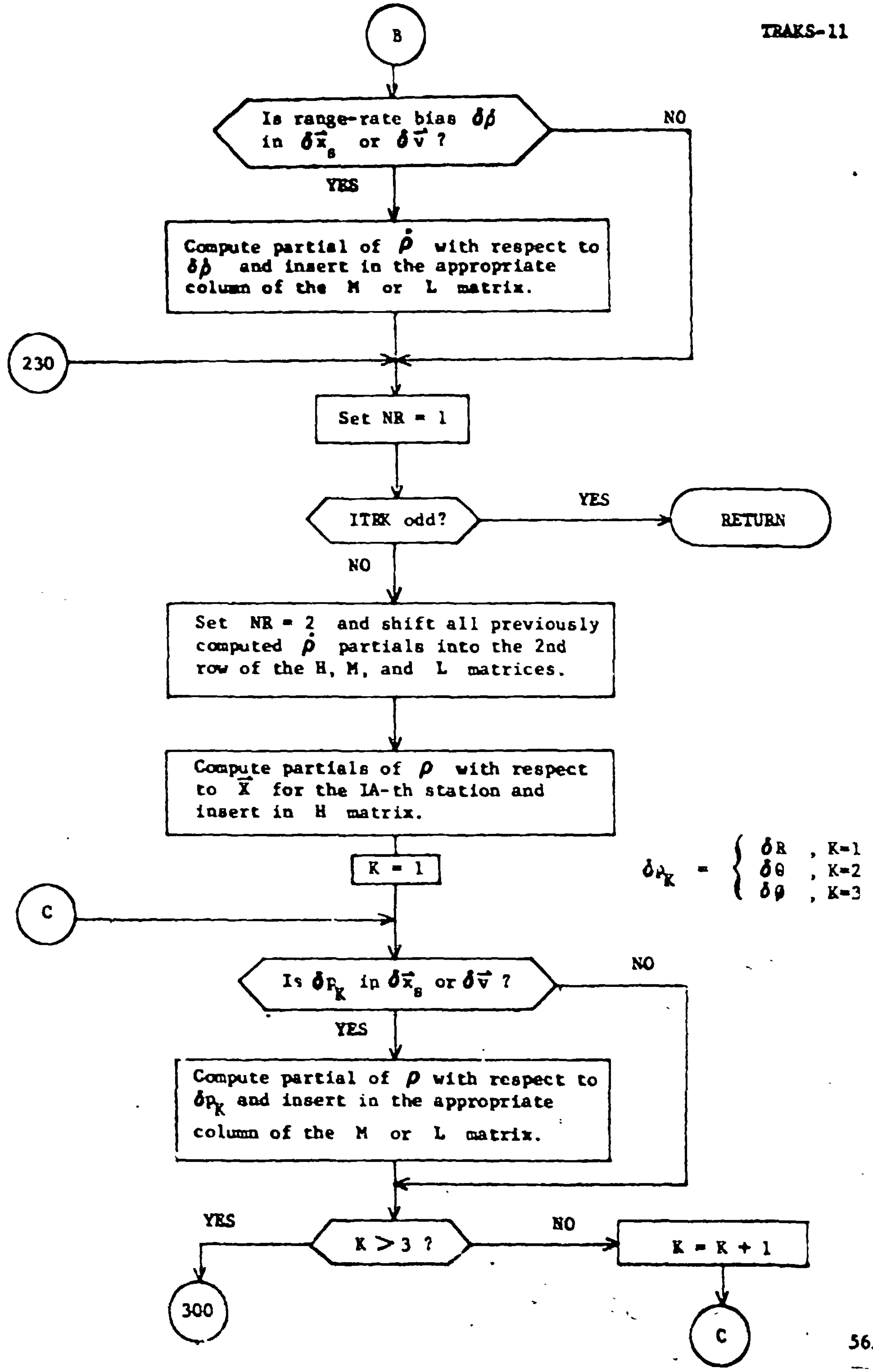
TRAKS Flow Chart

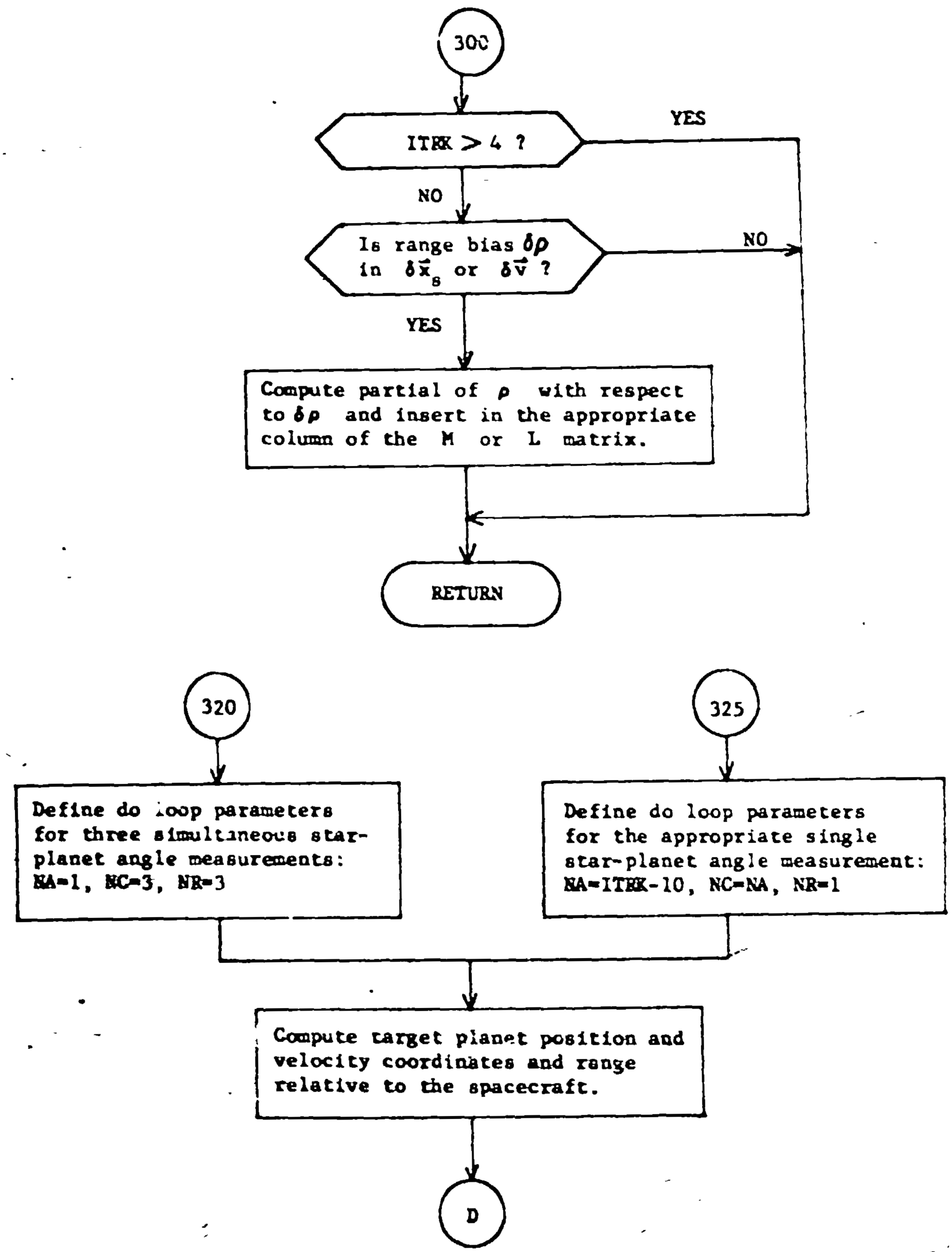


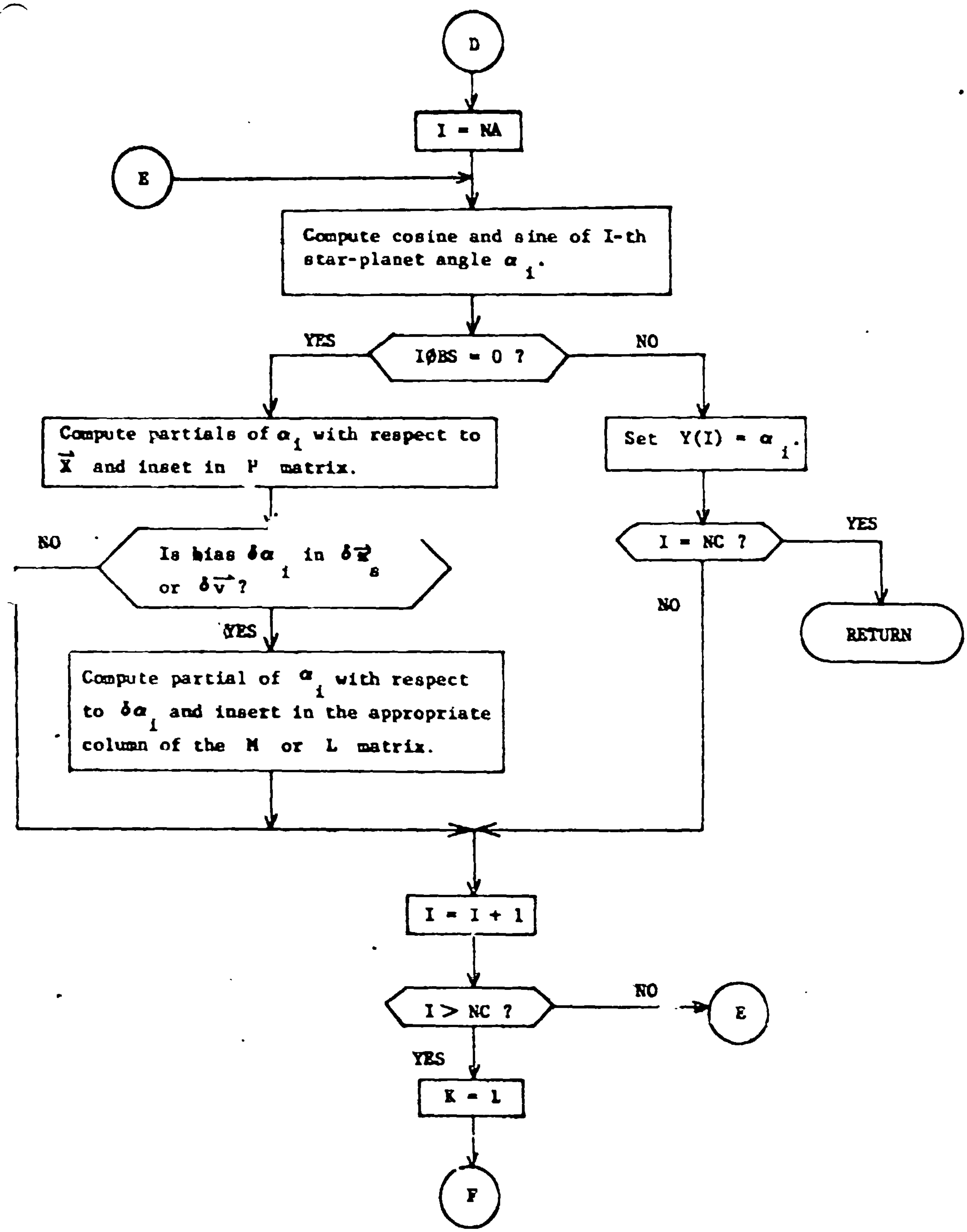

50
Compute spacecraft position and velocity coordinates relative to the Earth.
Compute range ρ and range-rate $\dot{\rho}$ of spacecraft relative to the Earth.
IØBS = 0 ?
YES
NO
ITRK = 1 ?
YES
NO
Set $Y = \dot{\rho}$ and NR = 1 .
Set $Y = [\rho , \dot{\rho}]^T$ and NR = 2 .
RETURN
Compute partials of $\dot{\rho}$ with respect to $\vec{X}$ for the idealized station and insert in H matrix. Set NR = ITRK.
ITRK = 1 ?
YES
RETURN
NO
Compute partials of ρ with respect to $\vec{X}$ for the idealized station and insert in H matrix.
RETURN


110
Compute nominal altitude, latitude, and longitude of the IA-th tracking station.
IØBS ≠ 2 ?
YES
NO
Compute actual altitude, latitude, and longitude of the IA-th tracking station.
Compute geocentric equatorial components of tracking station position and velocity relative to the center of the Earth.
Compute geocentric ecliptic components of tracking station position and velocity relative to the Earth.
Compute spacecraft position, velocity, range ρ , and range-rate $\dot{\rho}$ relative to IA-th tracking station.
IØBS = 0 ?
YES
150
NO
ITRK odd?
YES
NO
Set Y = $\dot{\rho}$ and NR = 1 .
Set Y = $[\rho, \dot{\rho}]^T$ and NR = 2 .
RETURN

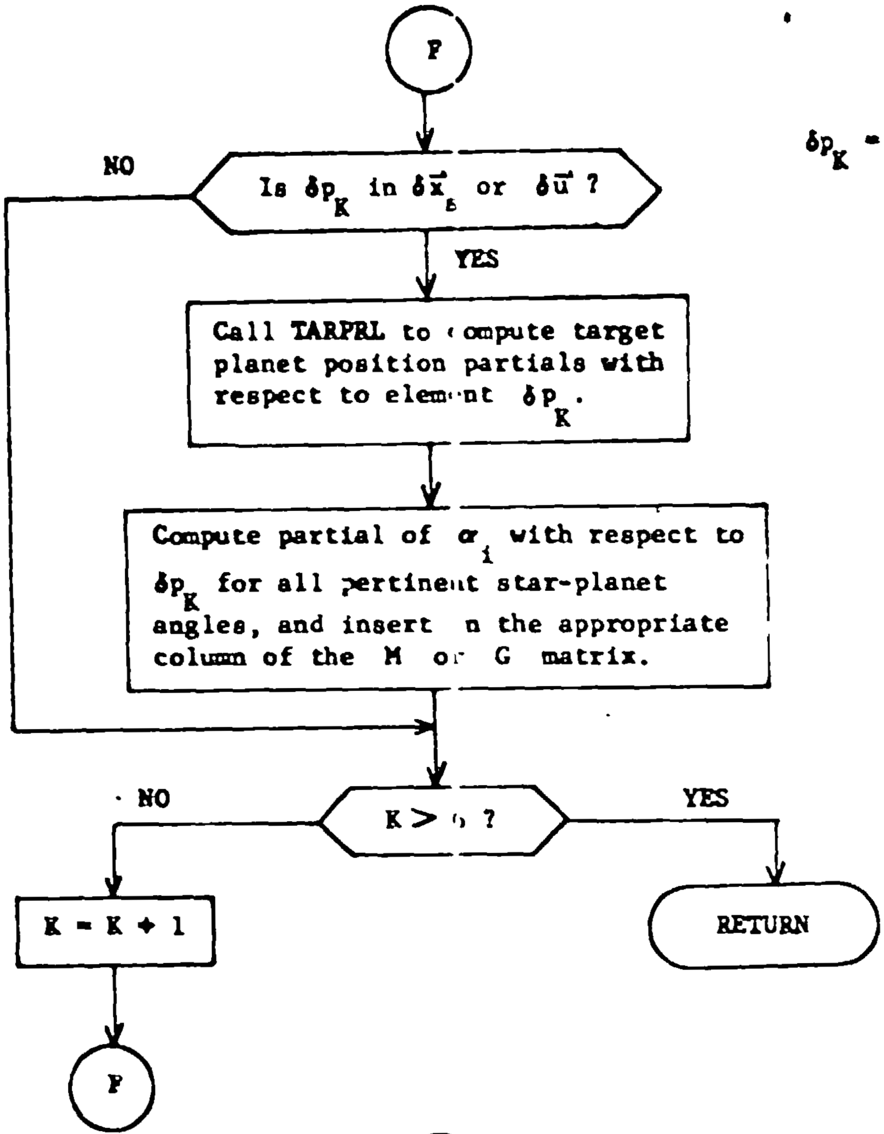

150
Compute partials of $\dot{\rho}$ with respect to $\overline{X}$ for the IA-th station and insert in H matrix.
Are any station location errors in $\delta\vec{x}_s$ or $\delta\vec{V}$?
NO
210
YES
Call STAPRL to compute partials of station position and velocity components with respect to station radius R , latitude θ , and longitude φ .
K = 1
$\delta p_K = \begin{cases} \delta R, & K=1 \\ \delta\theta, & K=2 \\ \delta\phi, & K=3 \end{cases}$
A
Is δp_K in $\delta\vec{x}_s$ or $\delta\vec{V}$?
NO
YES
Compute partial of $\dot{\rho}$ with respect to δp_K and insert in the appropriate column of the M or L matrix.
YES
K > 3 ?
NO
YES
ITRK > 4 ?
NO
K = K + 1
220
B
A

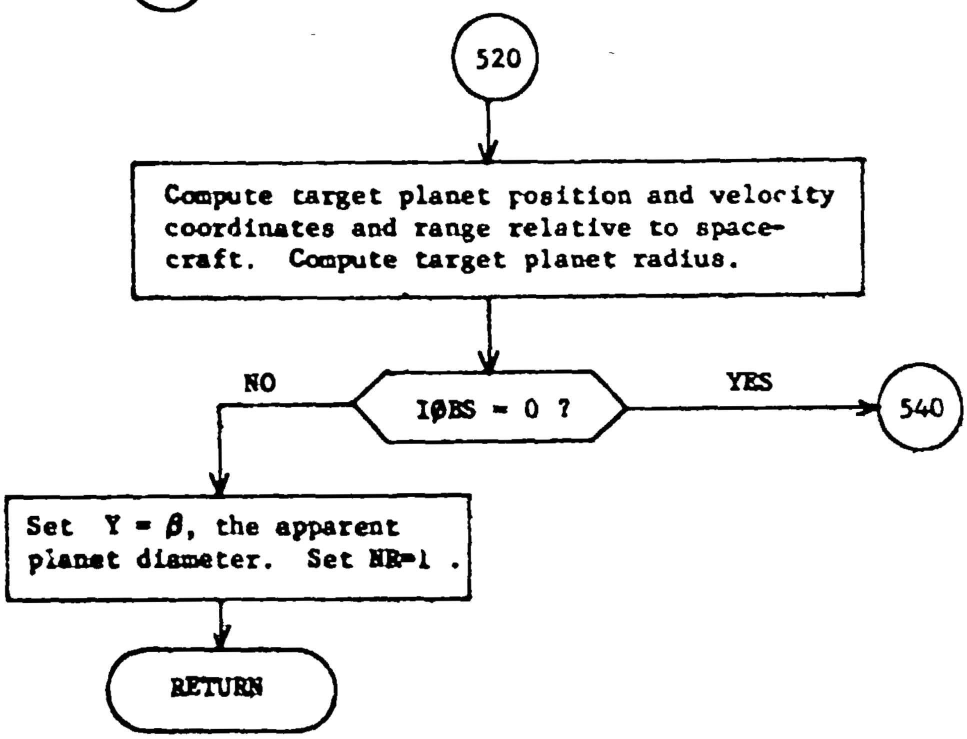

B
Is range-rate bias δρ̇ in δx̄s or δv̄ ?
NO
YES
Compute partial of ρ̇ with respect to δρ̇ and insert in the appropriate column of the M or L matrix.
230
Set NR = 1
ITRK odd?
YES
RETURN
NO
Set NR = 2 and shift all previously computed ρ̇ partials into the 2nd row of the H, M, and L matrices.
Compute partials of ρ with respect to X̄ for the IA-th station and insert in H matrix.
K = 1
δρK = δR , K=1; δθ , K=2; δφ , K=3
C
Is δρK in δx̄s or δv̄ ?
NO
YES
Compute partial of ρ with respect to δρK and insert in the appropriate column of the M or L matrix.
YES
K > 3 ?
NO
K = K + 1
300
C

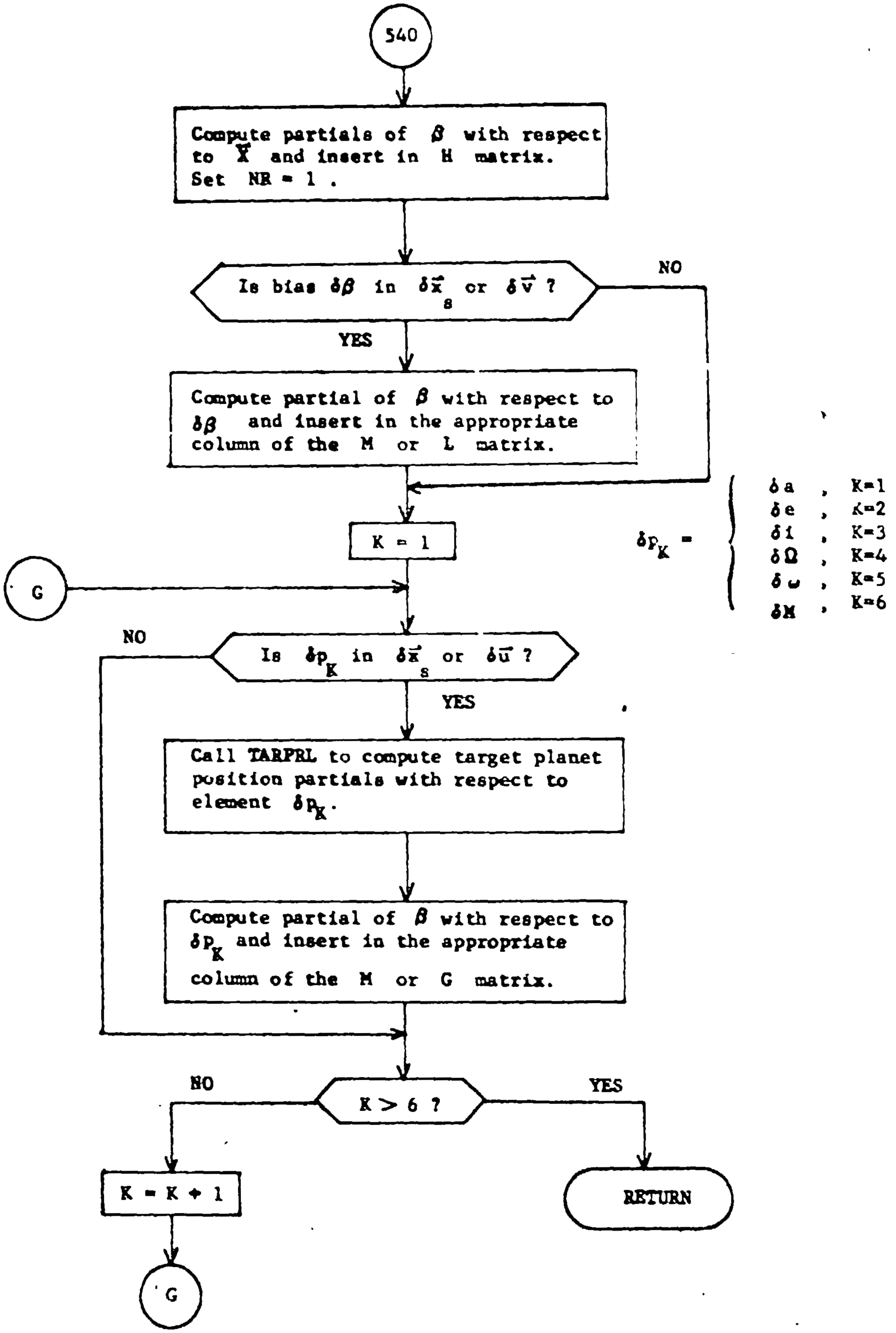

300
ITRK > 4 ?
YES
NO
Is range bias $\delta\rho$ in $\delta\vec{x}_s$ or $\delta\vec{v}$?
NO
YES
Compute partial of ρ with respect to $\delta\rho$ and insert in the appropriate column of the M or L matrix.
RETURN
320
325
Define do loop parameters for three simultaneous star-planet angle measurements: NA=1, NC=3, NR=3
Define do loop parameters for the appropriate single star-planet angle measurement: NA=ITRK-10, NC=NA, NR=1
Compute target planet position and velocity coordinates and range relative to the spacecraft.
D


D
I = NA
E
Compute cosine and sine of I-th star-planet angle α_i.
YES
IØBS = 0 ?
NO
Compute partials of α_i with respect to $\vec{X}$ and inset in P matrix.
Set Y(I) = α_i.
YES
I = NC ?
NO
RETURN
NO
Is bias $\delta\alpha_i$ in $\delta\vec{x}_s$ or $\delta\vec{v}$?
YES
Compute partial of α_i with respect to $\delta\alpha_i$ and insert in the appropriate column of the M or L matrix.
I = I + 1
I > NC ?
NO
E
YES
K = 1
F

$$\delta p_K = \begin{cases} \delta a , & K=1 \\ \delta e , & K=2 \\ \delta i , & K=3 \\ \delta \Omega , & K=4 \\ \delta \omega , & K=5 \\ \delta M , & K=6 \end{cases} .$$






540
Compute partials of β with respect to X̄ and insert in H matrix. Set NR = 1 .
Is bias δβ in δx⃗s or δv⃗ ?
NO
YES
Compute partial of β with respect to δβ and insert in the appropriate column of the M or L matrix.
K = 1
δpK = δa , K=1; δe , K=2; δi , K=3; δΩ , K=4; δω , K=5; δM , K=6
G
NO
Is δpK in δx⃗s or δu⃗ ?
YES
Call TARPRL to compute target planet position partials with respect to element δpK.
Compute partial of β with respect to δpK and insert in the appropriate column of the M or G matrix.
NO
K > 6 ?
YES
K = K + 1
RETURN
G

SUBROUTINE TRANS

PURPOSE: TO PERFORM ONE OF THE FOLLOWING THREE OPTIONS.

1. CONVERT FROM GEOCENTRIC EQUATORIAL RECTANGULAR COORDINATES TO GEOCENTRIC ELIPTIC COORDINATES
2. CONVERT FROM GEOCENTRIC EQUATORIAL COORDINATES TO HELIOCENTRIC ECLIPTIC COORDINATES
3. CONVERT FROM GEOCENTRIC ECLIPTIC COORDINATES TO HELIOCENTRIC ECLIPTIC COORDINATES

CALLING SEQUENCE: CALL TRANS(ICODE,X,Y,Z,VX,VY,VZ,XE,YE,ZE,VXE,VYE,VZE,EPS,ICODE2)

ARGUMENT:

EPS	I	OBLIQUITY OF EARTH
ICODE	I	AN INTERNAL CODE THAT DETERMINES IF OPTION 1 OR 2 ABOVE WILL BE EXERCISED
ICODE2	I	AN INTERNAL CODE THAT DETERMINES IF OPTION 3 ABOVE IS TO BE EXERCISED
VX	I/O	X-VELOCITY COMPONENT OF THE VEHICLE
VXE	I	X-VELOCITY COMPONENT OF EARTH IN HELIOCENTRIC ECLIPTIC COORDINATES
VY	I/O	Y-VELOCITY COMPONENT OF THE VEHICLE
VYE	I	Y-VELOCITY COMPONENT OF EARTH
VZ	I/O	Z-VELOCITY COMPONENT OF THE VEHICLE
VZE	I	Z-VELOCITY COMPONENT OF EARTH
X	I/O	X-POSITION COMPONENT OF THE VEHICLE
XE	I	X-POSITION COMPONENT OF THE EARTH IN HELIOCENTRIC ECLIPTIC COORDINATES
Y	I/O	Y-POSITION COMPONENT OF THE VEHICLE
YE	I	Y-POSITION COMPONENT OF THE EARTH
Z	I/O	Z-POSITION COMPONENT OF THE VEHICLE
ZE	I	Z-POSITION COMPONENT OF THE EARTH

SUBROUTINES SUPPORTED: DATA DATAS

LOCAL SYMBOLS:

CE	COSINE OF OBLIQUITY OF EARTH
DUM	INTERMEDIATE VARIABLE

SE SINE OF OBLIQUITY OF EARTH

TRANS Analysis

Subroutine TRANS transforms the position and velocity components of the spacecraft from one coordinate system to another. The three options available with this subroutine are summarized below.

1) Convert from geocentric equatorial coordinates to geocentric ecliptic coordinates using the following equations:

$$X = X$$
$$Y = Y \cos \epsilon + Z \sin \epsilon$$
$$Z = -Y \sin \epsilon + Z \cos \epsilon$$

$$\dot{X} = \dot{X}$$
$$\dot{Y} = \dot{Y} \cos \epsilon + \dot{Z} \sin \epsilon$$
$$\dot{Z} = -\dot{Y} \sin \epsilon + \dot{Z} \cos \epsilon$$

2) Convert from geocentric equatorial coordinates to heliocentric ecliptic coordinates. The same procedure as above is used to convert from geocentric equatorial to geocentric ecliptic. Then translate according to the following equations:

$$X = X + X_E \qquad \dot{X} = \dot{X} + \dot{X}_E$$
$$Y = Y + Y_E \qquad \dot{Y} = \dot{Y} + \dot{Y}_E$$
$$Z = Z + Z_E \qquad \dot{Z} = \dot{Z} + \dot{Z}_E$$

3) Convert from geocentric ecliptic coordinates to heliocentric ecliptic coordinates using the following equations:

$$X = X + X_E \qquad \dot{X} = \dot{X} + \dot{X}_E$$
$$Y = Y + Y_E \qquad \dot{Y} = \dot{Y} + \dot{Y}_E$$
$$Z = Z + Z_E \qquad \dot{Z} = \dot{Z} + \dot{Z}_E$$

SUBROUTINE TRAPAR

PURPOSE: TO COMPUTE THE FOLLOWING SET OF NAVIGATION PARAMETERS -- FLIGHT PATH ANGLE, ANGLE BETWEEN RELATIVE VELOCITY AND PLANE OF THE SKY, GEOCENTRIC DECLINATION, EARTH/SPACECRAFT/TARGET PLANET ANGLE, ANTENNA AXIS/LIMB OF SUN ANGLE, AND SPACECRAFT OCCULTATION RATIOS FOR SUN, MOON, AND PLANETS.

CALLING SEQUENCE: CALL TRAPAR

SUBROUTINES SUPPORTED: PRINT PRINT4 SETVEVS PRINT3 SETEVN

SUBROUTINES REQUIRED: EPHEM ORB PECEQ

LOCAL SYMBOLS:

ALFA	VECTOR FORMING RIGHT-HANDED ORTHOGONAL TRIAD WITH XN AND SSS VECTORS FOR CALCULATION OF ANTENNA AXIS/LIMB OF ANGLE OF SUN
AMAG	MAGNITUDE OF THE ALFA VECTOR
BETA	ANTENNA AXIS/EARTH ANGLE
CD	COSINE OF GEOCENTRIC DECLINATION
CT	INTERMEDIATE VARIABLE FOR ALL CALCULATIONS
CZAE	COSINE OF EARTH/SPACECRAFT/TARGET PLANE ANGLE
DELTA	GEOCENTRIC DECLINATION
DS	INTERMEDIATE VARIABLE FOR CALCULATION OF OCCULTATION RATIOS
ECEQP	TRANSFORMATION FROM EARTH ECLIPTIC TO EQUATORIAL FRAME FOR CALCULATION OF GEOCENTRIC DECLINATION
GAMMA	INERTIAL FLIGHT PATH ANGLE
IND	LOCATION IN THE F ARRAY OF THE EARTH POSITION AND VELOCITY IN THE INERTIAL FRAME
ISAVE	SAVES AND RESTORES FIRST ELEMENT OF THE NO-ARRAY FOR BARYCENTRIC NAVIGATION
JND	LOCATION IN THE F ARRAY OF THE TARGET PLANET POSITION AND VELOCITY IN THE INERTIAL FRAME

NINETY	CONSTANT VALUE, EQUAL TO 90.000
OCCULT	OCCULTATION RATIO OF THE I-TH PLANET
PHI	ANTENNA AXIS/LIMB OF SUN ANGLE
RDV	INTERMEDIATE VARIABLE, DOT PRODUCT OF TWO VECTORS
REMAG	MAGNITUDE OF THE EARTH HELIOCENTRIC POSITION
RIMAG	MAGNITUDE OF THE POSITION OF THE I-TH PLANET IN THE GEOCENTRIC ECLIPTIC FRAME
RMAG	MAGNITUDE OF THE SPACECRAFT HELIOCENTRIC POSITION
RSS	SPACECRAFT HELIOCENTRIC POSITION
SD	SINE OF GEOCENTRIC DECLINATION
SKYI	ANGLE BETWEEN SPACECRAFT VELOCITY RELATIVE TO EARTH AND PLANE OF THE SKY
SRDV	DOT PRODUCT OF SPACECRAFT GEOCENTRIC POSITION AND VELOCITY VECTORS
SRE	SPACECRAFT GEOCENTRIC ECLIPTIC POSITION AND VELOCITY
SRMAG	MAGNITUDE OF SPACECRAFT GEOCENTRIC POSITION
SRQ	SPACECRAFT GEOCENTRIC EQUATORIAL POSITION
SRTMAG	MAGNITUDE OF SRTP VECTOR
SRTP	SPACECRAFT ECLIPTIC POSITION RELATIVE TO TARGET PLANET
SVMAG	MAGNITUDE OF SPACECRAFT GEOCENTRIC VELOCITY
SX	INTERMEDIATE VARIABLE FOR ALL CALCULATIONS
SZAE	SINE OF EARTH/SPACECRAFT/TARGET PLANET ANGLE
THETA	INTERMEDIATE ANGLE USED TO CALCULATE NAVIGATION PARAMETERS
VMAG	MAGNITUDE OF SPACECRAFT VELOCITY RELATIVE

		TO INERTIAL FRAME
	XMAG	MAGNITUDE OF THE XN VECTOR BEFORE UNITIZING
	XN	CROSS PRODUCT OF SPACECRAFT GEOCENTRIC POSITION AND SPACECRAFT SPIN AXIS
	ZAE	EARTH/SPACECRAFT/TARGET PLANET ANGLE

COMMON COMPUTED/USED:	B	NO	RE		
COMMON USED:	F	IBARY	NBOD	NB	NTP
	ONE	PLANET	RADIUS	RAD	SSS
	TWO	V	XP	ZERO	

TRAPAR Analysis

The coordinate systems and variables required for the derivation of the first four navigation parameters are shown in Figure 1. The inertial coordinate system XYZ may be heliocentric or barycentric ecliptic. The position and velocity of the earth in inertial space is given by $\vec{r}_E$ and $\vec{v}_E$; that of the spacecraft, by $\vec{r}$ and $\vec{v}$; and that of the target planet (or moon), by $\vec{r}_{TP}$ and $\vec{v}_{TP}$. The xyz coordinate system is geocentric equatorial.

1. Flight path angle, γ.

   Let θ denote the angle between $\vec{r}$ and $\vec{v}$, so that

$$\cos\theta = \frac{\vec{r}\cdot\vec{v}}{r\,v} \quad \text{and} \quad \sin\theta = +\left[1-\cos^2\theta\right]^{\frac{1}{2}} .$$

   Then

$$\gamma = \frac{\pi}{2} - \theta .$$

2. Angle between relative velocity and plane of the sky, i'.

   The plane of the sky is defined as the plane perpendicular to the vector $\vec{r}-\vec{r}_E$. Let θ' denote the angle between $\vec{r}-\vec{r}_E$ and $\vec{v}-\vec{v}_E$, so that

$$\cos\theta' = \frac{(\vec{r}-\vec{r}_E)\cdot(\vec{v}-\vec{v}_E)}{\left|\vec{r}-\vec{r}_E\right|\cdot\left|\vec{v}-\vec{v}_E\right|} \quad \text{and} \quad \sin\theta' = +\left[1-\cos^2\theta'\right]^{\frac{1}{2}}$$

   Then

$$i' = \frac{\pi}{2} - \theta' .$$

   Note that i' is not defined if the relative velocity $\vec{v}-\vec{v}_E$ is zero.

3. Geocentric declination, δ.

   Let (x, y, z) denote the geocentric equatorial components of $\vec{r}-\vec{r}_E$. Then

$$\delta = \tan^{-1}\left(\frac{z}{\sqrt{x^2+y^2}}\right) .$$

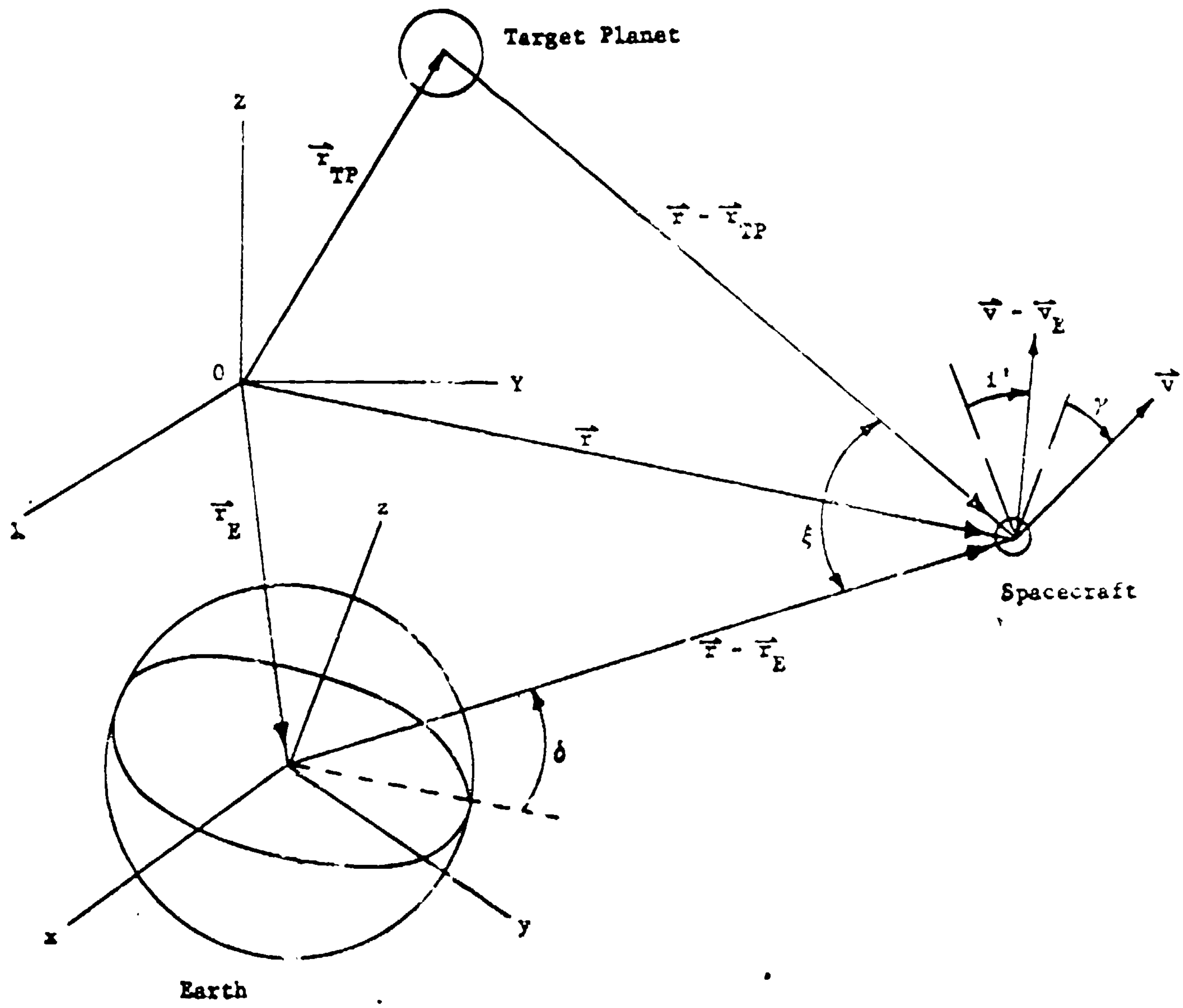

Target Planet
Z
$\vec{r}_{TP}$
$\vec{r} - \vec{r}_{TP}$
$\vec{v} - \vec{v}_E$
O
Y
i'
γ
$\vec{v}$
$\vec{r}$
X
$\vec{r}_E$
z
ξ
Spacecraft
$\vec{r} - \vec{r}_E$
δ
x
y
Earth

Figure 1

4. Earth/spacecraft/target planet angle, ξ.

The angle ξ is the angle between the vectors $\vec{r} - \vec{r}_E$ and $\vec{r} - \vec{r}_{TP}$, so that

$$\cos \xi = \frac{(\vec{r} - \vec{r}_E)\cdot(\vec{r} - \vec{r}_{TP})}{|\vec{r} - \vec{r}_E|\cdot|\vec{r} - \vec{r}_{TP}|}$$

and

$$\sin \xi = + \left[1 - \cos^2 \xi\right]^{\frac{1}{2}} .$$

The next two navigation parameters relate to the spacecraft antenna axis. The pertinent geometry is shown in Figure 2. The antenna axis $\vec{\alpha}$ is defined as the intersection between the antenna plane (the plane perpendicular to the spacecraft spin axis $\vec{s}$) and the plane formed by the $\vec{r} - \vec{r}_E$ and $\vec{s}$ vectors. The vector $\vec{\rho}$ originates from the limb of the sun and lies in the $\vec{r}$, $\vec{\alpha}$ plane.

5. Antenna axis/Earth angle, β.

Let ψ denote the angle between the unit spin axis vector $\vec{s}$ and $\vec{r} - \vec{r}_E$, so that

$$\cos \psi = \frac{\vec{s}\cdot(\vec{r} - \vec{r}_E)}{|\vec{r} - \vec{r}_E|} \quad \text{and} \quad \sin \psi = + \left[1 - \cos^2 \psi\right]^{\frac{1}{2}} .$$

Then

$$\beta = \frac{\pi}{2} - \psi .$$

Note that the antenna axis is not uniquely defined when the angle $\psi = 0$.

C7

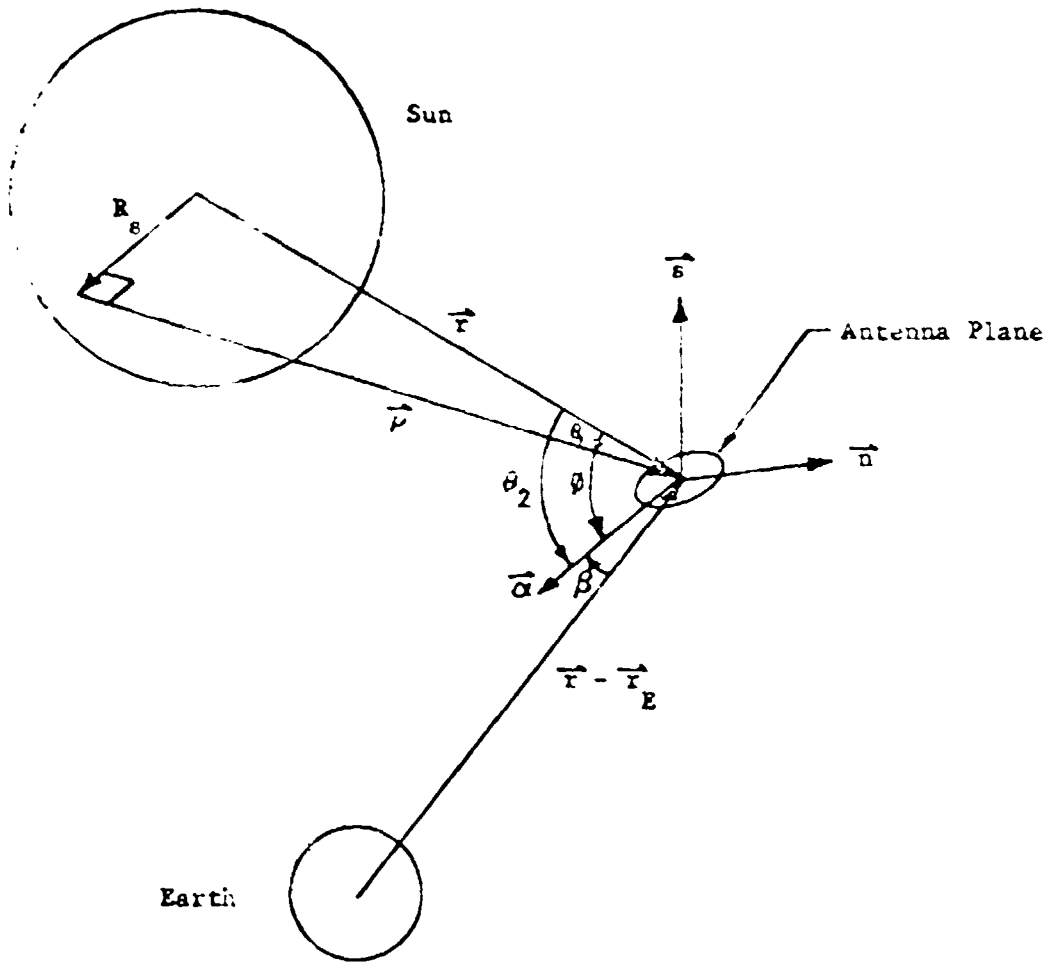


Figure 2. Antenna Axis Geometry

...a axis/limb of Sun angle, ϕ.

...nit vector $\vec{n}$ normal to the $\vec{s}$, $\vec{r} - \vec{r}_E$ plane is given by

$$\vec{n} = \frac{(\vec{r} - \vec{r}_E) \times \vec{s}}{|(\vec{r} - \vec{r}_E) \times \vec{s}|} .$$

...e unit antenna axis vector $\vec{\alpha}$ is given by

$$\vec{\alpha} = \vec{n} \times \vec{s} .$$

The angle θ_2 denotes the angle between the vectors $\vec{r}$ and $\vec{\alpha}$, so that

$$\cos\theta_2 = -\frac{\vec{r}\cdot\vec{\alpha}}{r} \quad \text{and} \quad \sin\theta_2 = +\left[1 - \cos^2\theta_2\right]^{\frac{1}{2}} .$$

The angle θ_1 denotes the angle between the vectors $\vec{\rho}$ and $\vec{r}$, so that

$$\theta_1 = \sin^{-1}\left(\frac{R_s}{r}\right), \quad 0 \leq \theta_1 \leq \frac{\pi}{2}$$

where R_s is the radius of the Sun.

Then

$$\phi = \theta_2 - \theta_1 .$$

The final set of navigation parameters relate to spacecraft occultation ratios for the Sun and all other celestial bodies assumed in the dynamic model. The pertinent geometry is shown in Figure 3. The position of the i-th celestial body relative to the Sun is denoted by $\vec{r}_i$. Occultation parameters d_s and d_i are defined as the minimal distances from the centers of the Sun and i-th body, respectively, to the Earth/spacecraft vector $\vec{r} - \vec{r}_E$.

7. Spacecraft occultation ratio for the Sun.

The occultation ratio for the Sun is defined as d_s/R_s, where R_s is the Sun radius. As long as the occultation ratio is greater than one, the spacecraft is neither being occulted by the Sun nor passing in front of the Sun. The occultation ratio is computed only when the angle between the $\vec{r} - \vec{r}_E$ and $\vec{r}_E$ vectors is less than or equal to 90 degrees, or, equivalently, when

$$\vec{r}_E \cdot (\vec{r} - \vec{r}_E) \leq 0.$$

If this condition is satisfied, the occultation ratio is computed using the equations

$$d_s = \left[r_E^{\,2} - b^2\right]^{\frac{1}{2}}$$

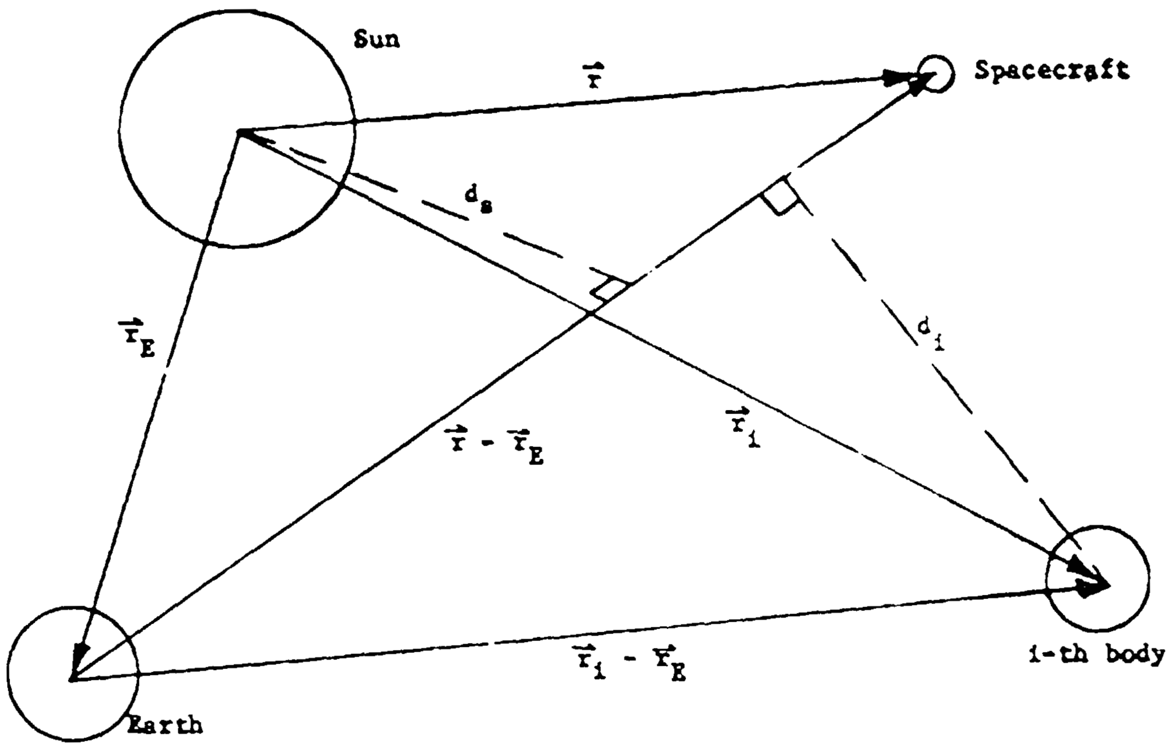


Figure 3. Occultation Geometry

and

$$b = \frac{-\vec{r}_E \cdot (\vec{r} - \vec{r}_E)}{\left|\vec{r} - \vec{r}_E\right|} .$$

Occultation occurs if $\frac{d_s}{R_s} \leq 1$ and $\left|\vec{r} - \vec{r}_E\right| \geq r_E$; if $\frac{d_s}{R_s} \leq 1$ and $\left|\vec{r} - \vec{r}_E\right| < r_E$, then the spacecraft is passing in front of the Sun.

8. Spacecraft occultation ratios for other celestial bodies.

The occultation ratio for the i-th celestial body is defined as d_i/R_i, where R_i is the radius of the i-th body. The occultation ratio is

computed only when

$$(\vec{r} - \vec{r}_E)\cdot(\vec{r}_i - \vec{r}_E) \geq 0 \quad .$$

If this conditions is satisfied, the occultation ratio is computed using the equations

$$d_i = \left[a_i^{\,2} - b_i^{\,2}\right]^{\frac{1}{2}}$$

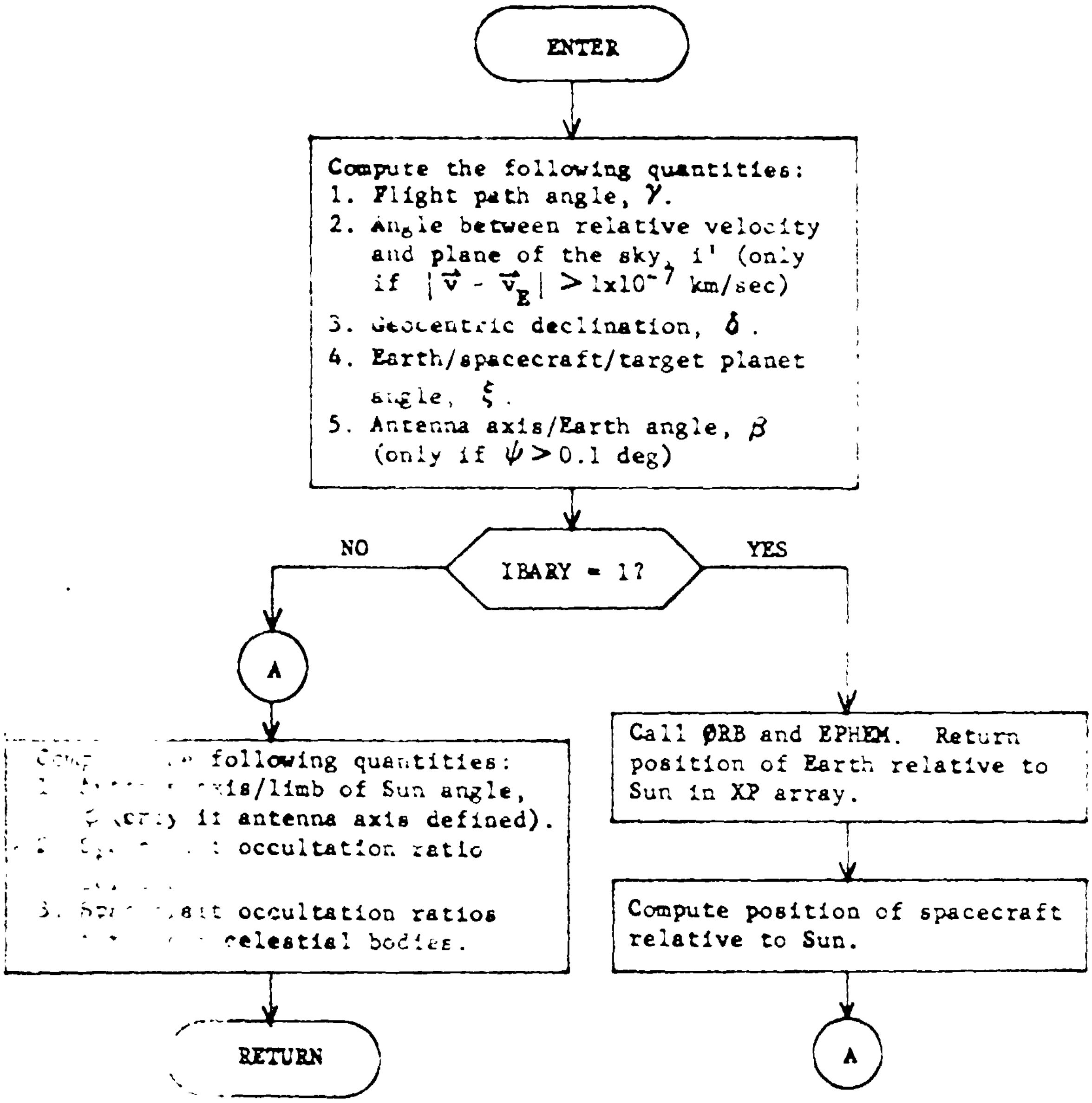
$$a_i = \left|\vec{r}_i - \vec{r}_E\right|$$

and

$$b_i = \frac{(\vec{r} - \vec{r}_E)\cdot(\vec{r}_i - \vec{r}_E)}{\left|\vec{r} - \vec{r}_E\right|} \quad .$$

Occultation occurs if $\frac{d_i}{R_i} \leq 1$ and $\left|\vec{r} - \vec{r}_E\right| \geq \left|\vec{r}_i - \vec{r}_E\right|$; if $\frac{d_i}{R_i} \leq 1$ and $\left|\vec{r} - \vec{r}_E\right| < \left|\vec{r}_i - \vec{r}_E\right|$, then the spacecraft is passing is front of the i-th celestial body.

TRAPAR Flow Chart



SUBROUTINE TRJTRY

PURPOSE: TO DETERMINE THE TIME OF THE NEXT GUIDANCE EVENT AND INTEGRATE THE NOMINAL TRAJECTORY FROM THE PREVIOUS EVENT TIME TO THE NEXT TIME.

CALLING SEQUENCE: CALL TRJTRY

SUBROUTINES SUPPORTED: NOMNAL

SUBROUTINES REQUIRED: VMP

LOCAL SYMBOLS:

ACK	ACCURACY USED TO INTEGRATE THE NOMINAL TRAJECTORY
DELMIN	TIME(DAYS) BETWEEN THE LAST EVENT AND THE NEXT EVENT
DELTM	SAME AS DELMIN -THE TIME VMP IS TO INTEGRATE THE TRAJECTORY UNLESS ANOTHER STOPPING CONDITION OCCURS
DELT	SAME AS DELMIN
ERROR	MINIMUM ALLOWABLE VALUE OF DELMIN
ISP2	FLAG TO CONTROL STOPPING CONDITION =1 STOP AT SHPERE-OF-INFLUENCE =0 DO NOT STOP AT SHPERE-OF-INFLUENCE
I	INDEX
J	INDEX
RSF	SPACECRAFT STATE AT FINAL TIME

1

COMMON COMPUTED/USED: D1 ICL2 ICL ISPH KSICA KTIM RIN TIMG TRTM

COMMON COMPUTED: DELTP IEPHEM INPR IPRINT KUR

COMMON USED: ACKT KGYD NCPR NOGYD TMPR V

TRJTRY Analysis

TRJTRY determines the time of the next guidance event and integrates the nominal trajectory from the previous event time to the next time.

Special provisions must be made in determining the next guidance event because of the flexibility permitted in specifying the times of those guidance events. For every guidance event i, parameters KTIM(i) and TIMG(i) will have been set before entering TRJTRY. KTIM(i) prescribes the epoch to which the guidance event i is referenced with KTIM(i) = 1,2,3 corresponding to epochs of initial time, sphere of influence (SOI) intersection, and closest approach (CA) passage respectively. TIMG(i) then specifies the time interval (days) from the epoch to the guidance event. The guidance events do not need to be arranged chronologically. After execution of each guidance event i the flag KTIM(i) is set equal to 0.

The first computational procedure in TRJTRY is the sequencing loop. Here a search determines the minimum value of TIMG(i) over all values of i such that KTIM(i) = 1. The time interval Δt between that time and the current time is then computed. If Δt is less than an allowable tolerance ϵ ($=10^{-5}$ days) the program returns to NOMNAL for the processing of the current event.

If $\Delta t \geq \epsilon$ TRJTRY must perform an integration to the next guidance event. TRJTRY first sets up flags controlling integration stopping conditions depending upon the current value of KSICA. The flag KSICA determines the current phase of the trajectory. KSICA is initially set equal to 1 (PRELIM). When the target planet SOI is encountered KSICA is set to 2. Finally when CA to the target planet occurs it is set to 3.

The stopping condition flags are ISP2 and ICL2. The flag ISP2 determines whether the integration should be stopped at SOI if encountered (ISP2 = 1) or not (ISP2 = 0). The flag ICL2 determines whether the integration should be stopped at CA if encountered (ICL2 = 1) or not (ICL2 = 0).

Therefore if KSICA = 1, TRJTRY sets ISP2 = 1 so that the integration will stop at the guidance event time only if that time occurs before SOI. But if the SOI is encountered before the event time, all times referenced to the SOI must be updated before determining the next event. Similarly when KSICA = 2 TRJTRY sets ICL2 = 1 so that times referenced to CA may be updated when CA occurs. Of course when KSICA = 3, all times have been updated (referenced to initial time) and neither ISP2 nor ICL2 need be set to 1.

Having set the stopping condition flags, TRJTRY now calls VMP for the propagation of the trajectory to the required stopping condition. At the end of the integration it records the current trajectory time and state.

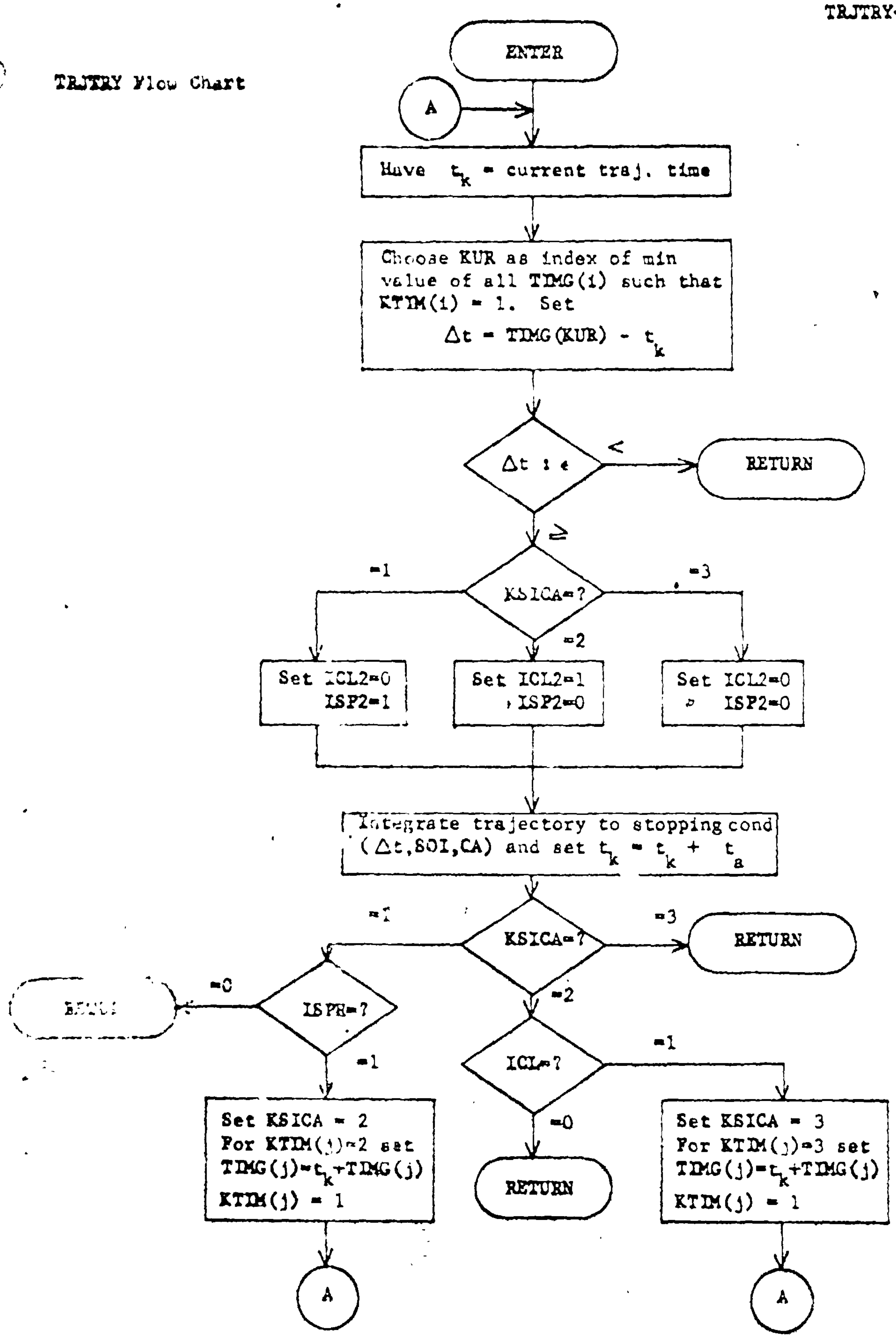
TRJTRY now sorts again on KSICA. If KSICA = 3, the trajectory has been integrated to the time of the current event and so control may be returned to NOMNAL.

If KSICA = 1 the SOI had not yet been reached at the previous event. TRJTRY then checks the flag ISPH. The flag ISPH reveals whether the current trajectory intersected the target planet SOI (ISPH = 1) or did not (ISPH = 0). Therefore if ISPH = 0, the current guidance event occurred before the trajectory intersected the SOI and thus the current state corresponds to the time of the guidance event. Therefore the return is made to NORMAL.

If however KSICA = 1 and ISPH = 1 the trajectory integration was stopped at the SOI. TRJTRY now sets KSICA = 2 and updates all times referenced to the SOI so that they are now referenced to initial time (KTIM(i) = 1). It reenters the sequencing loop to determine the time of the next guidance event where the candidate events now include those originally referenced to SOI.

Similar steps are made when KSICA = 2. The flag ICL designates whether the current trajectory had a CA (ICL = 1) or not (ICL = 0). If KSICA = 2 and ICL = 0, the trajectory encountered the guidance event before reaching a CA so the return is made to NORMAL. If KSICA = 2 and ICL = 1, the final time and state of the trajectory refer to closest approach. In this case TRJTRY sets KSICA = 3 and updates to initial time all times originally referenced to CA. It then returns to the sequencing loop.

TRJTRY Flow Chart



SUBROUTINE VARADA

PURPOSE COMPUTE VARIATION MATRIX FOR THREE-VARIABLE B-PLANE GUIDANCE POLICY IN THE ERROR ANALYSIS PROGRAM

CALLING SEQUENCE: CALL VARADA(RI,XSIP,XSIV,TEVN,TSI,ADA,BS,BOTS,BDRS)

ARGUMENT:

ADA	O	VARIATION MATRIX
BS	I	B OF THE NOMINAL TRAJECTORY
BDRS	I	B DOT R OF THE NOMINAL TRAJECTORY
BDTS	I	B DOT T OF THE NOMINAL TRAJECTORY
RI	I	POSITION AND VELOCITY OF THE VEHICLE AT THE TIME OF GUIDANCE EVENT
TEVN	I	TRAJECTORY TIME OF THE GUIDANCE EVENT
TSI	I	TRAJECTORY TIME AT WHICH THE VEHICLE REACHED THE SPHERE OF INFLUENCE ON THE NOMINAL TRAJECTORY
XSIP	I	POSITION OF THE VEHICLE AT THE SPHERE OF INFLUENCE ON THE NOMINAL TRAJECTORY
XSIV	I	VELOCITY OF THE VEHICLE AT THE SPHERE OF INFLUENCE ON THE NOMINAL TRAJECTORY

SUBROUTINES SUPPORTED: GUID

SUBROUTINES REQUIRED: NTM

LOCAL SYMBOLS:

BDR1	TEMPORARY STORAGE FOR BDR
BDT1	TEMPORARY STORAGE FOR BDT
B1	TEMPORARY STORAGE FOR B
DSI1	TEMPORARY STORAGE FOR DSI
IPR	TEMPORARY STORAGE FOR IPRINT
ISP	TEMPORARY STORAGE FOR ISP2
IPO	TEMPORARY STORAGE FOR IPRINT
RF	ALTERED FINAL STATE OF VEHICLE
TSI1	TEMPORARY STORAGE FOR TSI

XC	ALTERED INITIAL STATE OF VEHICLE				
COMMON COMPUTED/USED:	BDR ISP2	BDT	DSI	IPRINT	ISPH
COMMON COMPUTED:	B	DELTM	RSI	TRIM1	VSI
COMMON USED:	DATEJ NTP	FACP	FACV	FUTG	NTMC

VARADA Analysis

Subroutine VARADA employs numerical differencing to compute the variation matrix η for the three-variable B-plane guidance policy in the guidance event of the error analysis mode. See subroutine VARSIM Analysis for further analytical details, since the only difference between VARADA and VARSIM is that VARADA computations are based on the most recent targeted nominal, while VARSIM computations are based on the most recent nominal. The VARADA flow chart is identical to that of VARSIM except for the fact that in VARADA the nominal position/velocity state at t_{SI} is saved prior to calling VARADA, while in VARSIM it is saved locally.

SUBROUTINE VARSIM

PURPOSE COMPUTE VARIATION MATRIX FOR THREE-VARIABLE B-PLANE GUIDANCE POLICY IN THE SIMULATION PROGRAM

CALLING SEQUENCE: CALL VARSIM(RI1,TEVN,TSI,ADA)

ARGUMENT:	ADA	O	VARIATION MATRIX
	RI1	I	VEHICLE POSITION/VELOCITY ON MOST RECENT NOMINAL TRAJECTORY AT TIME OF THE GUIDANCE EVENT
	TEVN	I	TRAJECTORY TIME OF GUIDANCE EVENT
	TSI	I	TRAJECTORY TIME AT SPHERE OF INFLUENCE

SUBROUTINES SUPPORTED: GUISS

SUBROUTINES REQUIRED: NTMS

LOCAL SYMBOLS:	BDRS	TEMPORARY STORAGE FOR BDR
	BDTS	TEMPORARY STORAGE FOR BDT
	BS	TEMPORARY STORAGE FOR B
	IPR	TEMPORARY STORAGE FOR IPRINT
	ISPS	TEMPORARY STORAGE FOR ISP2
	RF1	ALTERED FINAL STATE OF VEHICLE ON MOST RECENT NOMINAL
	RSIS	TEMPORARY STORAGE FOR RSI
	TSI1	TEMPORARY STORAGE FOR TSI
	VSIS	TEMPORARY STORAGE FOR VSI
	XC	ALTERED INITIAL STATE OF VEHICLE ON MOST RECENT NOMINAL

COMMON COMPUTED/USED:	BDR	BDT	B	DSI	IPRINT
	ISPH	ISP2	RSI	VSI	
COMMON COMPUTED:	TRTM1				
COMMON USED:	DATEJ	FACP	FACV	NTMC	NTP

VARSIM Analysis

Subroutine VARSIM employs numerical differencing to compute the variation matrix η for the three-variable B-plane guidance policy in the guidance event of the simulation mode. This variation matrix relates deviations in the position/velocity state at t_k to deviations in B·T, B·R, and t_{SI}:

$$\begin{bmatrix} \delta B\cdot T \\ \delta B\cdot R \\ \delta t_{SI} \end{bmatrix} = \eta \begin{bmatrix} \delta\vec{R}_k \\ \delta\vec{V}_k \end{bmatrix} = \eta\delta\vec{X}_k$$

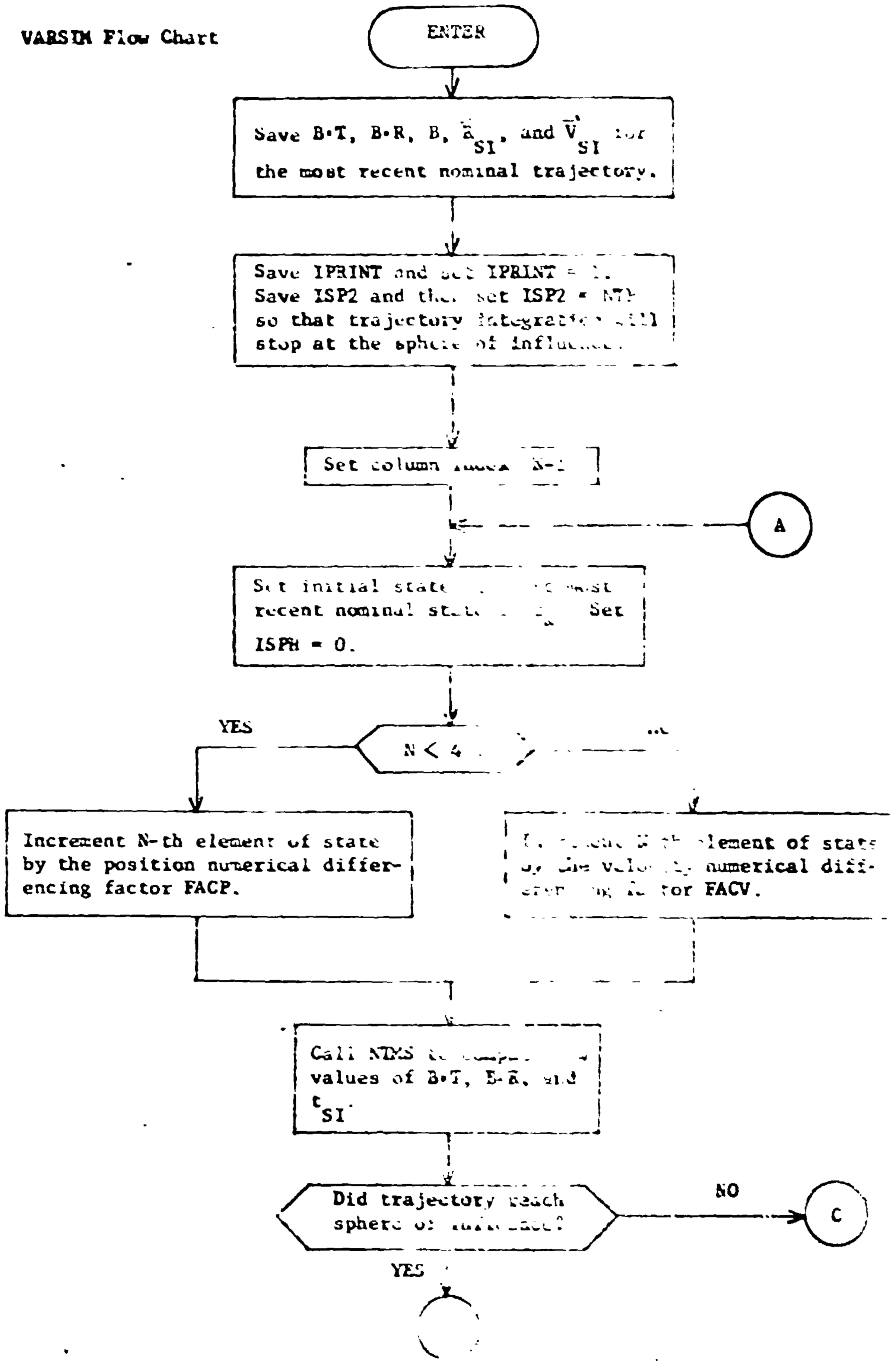
Since no good analytical formulas which relate δt_{SI} to $\delta\vec{R}_k$ and $\delta\vec{V}_k$ exist, numerical differencing must be employed to compute η.

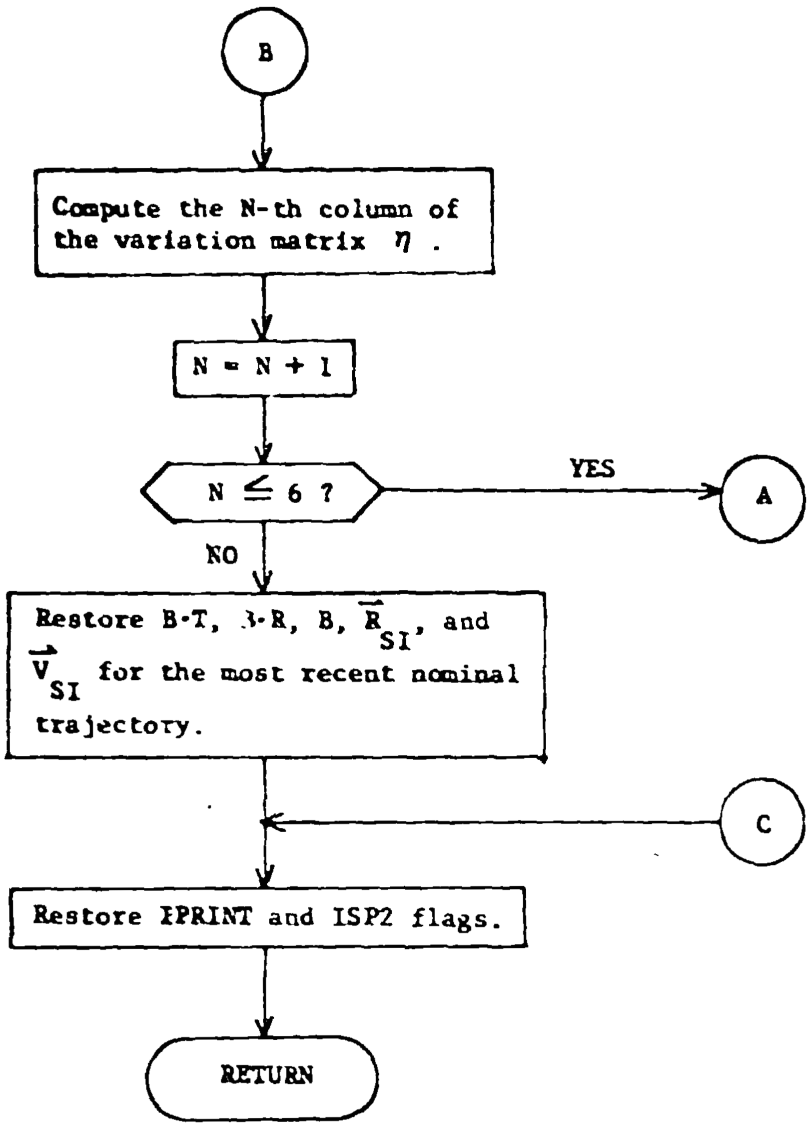
Let $\vec{\eta}_j$ be the j-th column of the matrix η, and assume (most recent) nominal $B\cdot T^*$, $B\cdot R^*$, t_{SI}^*, and $\vec{X}_k^*$ are available. To obtain $\vec{\eta}_j$ we increment the j-th element of $\vec{X}_k^*$ by the numerical differencing factor Δx_j and numerically integrate the spacecraft equations of motion from t_k to the sphere of influence of the target planet to obtain the new values of B·T, B·R, and t_{SI}. Then

$$\vec{\eta}_j = \left[\frac{B\cdot T - B\cdot T^*}{\Delta x_j} , \quad \frac{B\cdot R - B\cdot R^*}{\Delta x_j} , \quad \frac{t_{SI} - t_{SI}^*}{\Delta x_j} \right]^T$$

$$j = 1,2,\ldots, 6$$

VARSIM Flow Chart




B
Compute the N-th column of the variation matrix η .
N = N + 1
N $\leq$ 6 ?
YES
A
NO
Restore B·T, B·R, B, $\vec{R}_{SI}$, and $\vec{V}_{SI}$ for the most recent nominal trajectory.
C
Restore IPRINT and ISP2 flags.
RETURN

SUBROUTINE VECTOR

PURPOSE. TO COMPUTE THE VECTOR ORBITAL ELEMENTS K (ANGULAR MOMENTUM VECTOR), E (ECCENTRICITY VECTOR TOWARD PERIHELION), TO COMPUTE THE SPACECRAFT FINAL POSITION ON THE ORBIT TO ACCURATELY APPROXIMATE THE DESIRED TIME INTERVAL, AND TO COMPUTE THE CONIC SECTION TIME OF FLIGHT.

CALLING SEQUENCE: CALL VECTOR

SUBROUTINES SUPPORTED: VMP

SUBROUTINES REQUIRED: SPACE

LOCAL SYMBOLS: DUM INTERMEDIATE VARIABLE

COMMON COMPUTED/USED: V

COMMON COMPUTED: KOUNT

COMMON USED: HALF ITRAT ONE PI THREE TWOPI TWO

VECTØR Analysis

The Kepler vector $\vec{k}$ representing twice the areal rate of the spacecraft with respect to the virtual mass to be used during the current interval is computed from

$$\vec{k} = \vec{r}_{VS_B} \times \dot{\vec{r}}_{VS_B} \tag{1}$$

where the position and velocity vectors are referenced to the virtual mass at the beginning of the interval. The eccentricity vector for the interval is given by

$$\vec{e} = -\frac{\vec{r}_{VS_B}}{r_{VS_B}} - \frac{\vec{k} \times \dot{\vec{r}}_{VS_B}}{\bar{\mu}_V} \tag{2}$$

where $\bar{\mu}_V$ is the average value of the virtual mass during the interval.

The current time interval is computed from

$$\Delta\tau = \Delta t + K\,\Delta t^2 \tag{3}$$

where the factor K was precomputed during the previous iterations. The direction of the final position $\vec{\sigma}$ is determined from

$$\vec{\sigma} = \vec{r}_{VS_B} + \Delta\tau\, \dot{\vec{r}}_{VS_B} \tag{4}$$

The magnitude factor B is chosen to force the final position to satisfy the orbit equation ($\vec{e} \cdot \vec{r} = -r + k^2/\mu$)

$$B = \frac{k^2/\bar{\mu}_V}{\vec{e} \cdot \sigma + |\sigma|} \tag{5}$$

The position and velocity vectors of the spacecraft relative to the virtual mass at the end of the interval are then

$$\vec{r}_{VS_E} = B\vec{\sigma}$$

$$\dot{\vec{r}}_{VS_E} = \frac{\bar{\mu}_V}{k^2}\left[\vec{k} \times \left(\vec{e} + \frac{\vec{r}_{VS_E}}{r_{VS_E}}\right)\right] \tag{6}$$

The final position and velocity of the spacecraft in the reference inertial coordinates are computed from

$$\vec{r}_{S_E} = \vec{r}_{VS_E} + \vec{r}_{V_E}$$

$$\dot{\vec{r}}_{S_E} = \dot{\vec{r}}_{VS_E} + \dot{\vec{r}}_{V_E} \qquad (7)$$

The exact conic section time of flight is now computed. The in-plane normal to the major axis is

$$\vec{n} = \frac{\vec{k} \times \vec{e}}{k\,e} \qquad e \neq 0$$

$$\phantom{\vec{n}} = \frac{\vec{k} \times \vec{r}_{VS_B}}{k\, r_{VS_B}} \qquad e = 0 \qquad (8)$$

The length of the semi-major axis is given by

$$a = \frac{k^2}{\bar{\mu}_V \; |1-e^2|^{\frac{1}{2}}} \qquad e \neq 1$$

$$a_i = \frac{2}{r_{VS_i} - k^2/\bar{\mu}_V} \qquad e = 1,\ i = B,E \qquad (9)$$

The projection of the radius vector orthogonal to the major axis normalized by a is given by

$$X_i = \frac{\vec{n} \cdot \vec{r}_{VS_i}}{a_i} \qquad i = B,E \qquad (10)$$

The mean angular rate is

$$\bar{\omega} = \frac{\bar{\mu}_V (1-e^2)}{k a} \qquad e \neq 1$$

$$= \frac{k}{2} \qquad e = 1 \qquad (11)$$

where $\omega < 0$ for hyperbolic orbits. The eccentric anomaly is given by

$$
\begin{aligned}
\underset{i=B,E}{E_i} &= \sin^{-1} X_i && e < 1 \\
&= \frac{k^2/\bar{\mu}_V \quad X_i}{3} && e = 1 \qquad (12) \\
&= \sinh^{-1} X_i && e > 1
\end{aligned}
$$

Then $M_i = E_i - e\,X_i \qquad i = B,E$ (13)

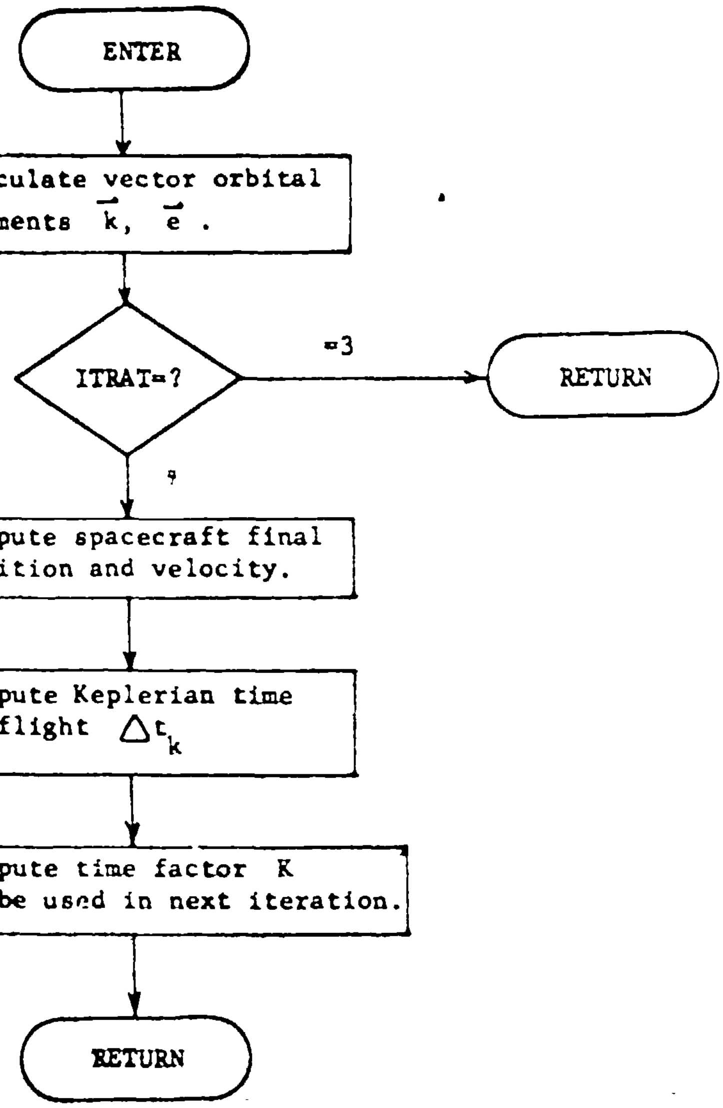
and the actual conic time of flight is

$$\Delta t = t_E - t_B = \frac{M_E - M_B}{\bar{\omega}} \tag{14}$$

The value of the time factor K to be used in the following interval is then computed

$$K = \frac{\Delta\tau - \Delta t}{(\Delta t)^2} \tag{15}$$

VECTOR Flow Chart



SUBROUTINE VMASS

PURPOSE: TO COMPUTE THE POSITION, VELOCITY, MAGNITUDE, AND MAGNITUDE RATE OF THE VIRTUAL MASS.

CALLING SEQUENCE: CALL VMASS

SUBROUTINES SUPPORTED: VMP

SUBROUTINES REQUIRED: NONE

COMMON COMPUTED/USED: F V

COMMON USED: NBODY THREE ZERO

VMASS Analysis

The current virtual mass data is computed by VMASS. The magnitude and position of the virtual mass is given by

$$\mu_V = r_{VS}^3 M_S \tag{1}$$

$$\vec{r}_V = \frac{\vec{M}}{M_S} \tag{2}$$

where the intermediate variables are given by

$$\vec{M} = \sum_{i=1}^{n} \frac{\mu_i \vec{r}_i}{r_{iS}^3} \tag{3}$$

$$M_S = \sum_{i=1}^{n} \frac{\mu_i}{r_{iS}^3} \tag{4}$$

and of course $r_{iS} = |\vec{r}_i - \vec{r}_S|$ and $\vec{r}_{VS} = |\vec{r}_V - \vec{r}_S|$ where $\vec{r}_i$ represents the inertial position vector of the i-th body.

The time derivatives of these variables are given by

$$\dot{\mu}_V = \mu_V (\alpha_V + \frac{\dot{M}_S}{M}) \tag{5}$$

$$\dot{\vec{r}}_V = \frac{\dot{\vec{M}} - \vec{r}_V \dot{M}_S}{M_S} \tag{6}$$

$$\dot{\vec{M}} = \sum_{i=1}^{n} \frac{\mu_i}{r_{iS}^3} \left[\dot{\vec{r}}_i - \vec{r}_i \alpha_i\right] \tag{7}$$

$$\dot{M}_S = -\sum \frac{\mu_i}{r_{iS}^3} \alpha_{iS} \tag{8}$$

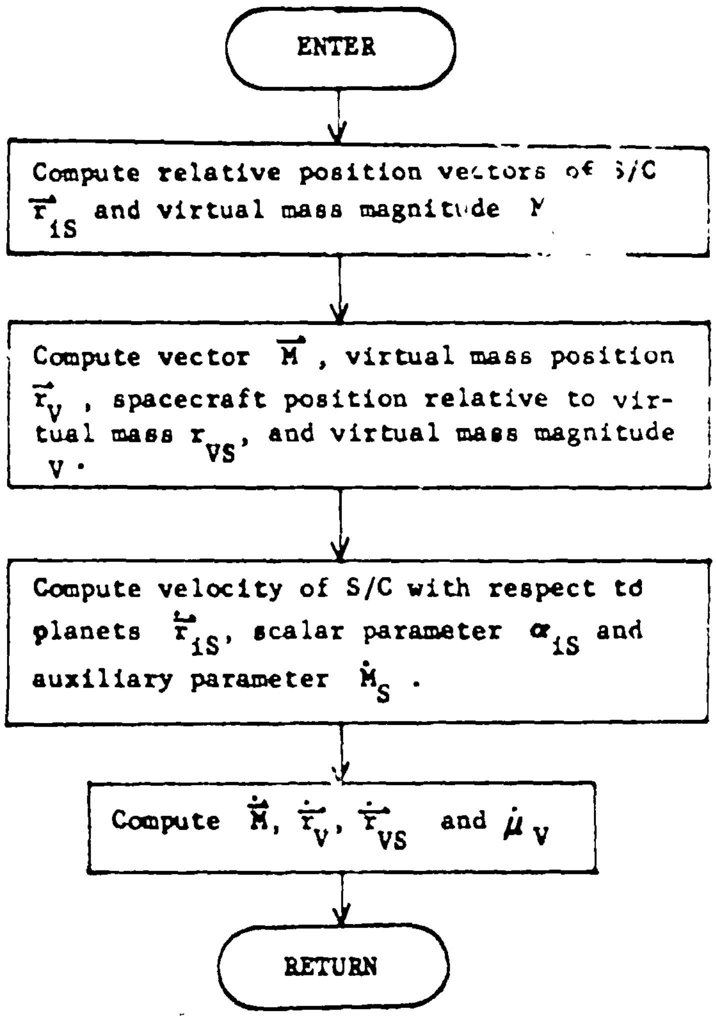
where

$$\alpha_{iS} = \frac{3\,\vec{r}_{iS}\cdot\dot{\vec{r}}_{iS}}{r_{iS}^{2}} \qquad (9)$$

Finally, the velocity of the spacecraft with respect to the virtual mass is

$$\dot{\vec{r}}_{VS} = \dot{\vec{r}}_{S} - \dot{\vec{r}}_{V} \qquad (10)$$

VMASS Flow Chart



SUBROUTINE VMP

PURPOSE PROVIDE LOGIC TO GENERATE VIRTUAL MASS TRAJECTORY

CALLING SEQUENCE: CALL VMP(RS,ACC,D1,TRTM,DELTM,RSF,ISP2)

ARGUMENTS	RS(6)	I	INERTIAL POSITION AND VELOCITY OF S/C AT INITIAL TIME
	ACC	I	ACCURACY USED IN INTEGRATION
	D1	I	JULIAN DATE, EPOCH 1900, OF INITIAL TIME
	TRTM	I	TRAJECTORY TIME (DAYS) AT INITIAL TIME
	DELTM	I	TIME INTERVAL IN DAYS OVER WHICH THE TRAJECTORY IS TO BE PROPAGATED UNLESS A STOPPING CONDITION IS REACHED
	RSF(6)	O	INERTIAL POSITION AND VELOCITY OF S/C AT FINAL TIME
	ISP2	I	SPHERE OF INFLUENCE STOPPING FLAG =0 DO NOT STOP AT SOI =1 STOP AT SOI IF INTERSECTED BEFORE FINAL TIME

SUBROUTINES SUPPORTED: CASCAD NTMS GIDANS TARGET TARMAX NTM TRJTRY DESENT

SUBROUTINES REQUIRED: CAREL ELCAR EPHEM IMPACT ORB PECEQ TIME ESTMT INPUTZ PRINT SPACE VECTOR VMASS

LOCAL SYMBOLS	AU	NOT USED
	CXI	COSINE OF THE TRAJECTORY INCLINATION AT CLOSEST APPROACH
	D	INTERMEDIATE DATE FOR PRINTOUT PURPOSES
	DELR	INTERMEDIATE VARIABLE FOR INTERSECTION OF SPHERE-OF-INFLUENCE
	DELT	INTERMEDIATE TIME INCREMENT FOR INTER-POLATED SPHERE-OF-INFLUENCE POSITION
	ECEQP	TRANSFORMATION FROM ECLIPTIC TO EQUATORIAL SYSTEM FOR TARGET PLANET
	ICUT	CUTOFF FLAG USED WHEN CLOSEST APPROACH CUTOFF WAS DESIRED BUT NO VALID CLOSEST

APPROACH FOUND

IDAY	PRINTOUT CALENDAR DAY
IHR	PRINTOUT CALENDAR HOUR
IMO	PRINTOUT CALENDAR MONTH
IP	NUMBER OF PLANET, USED IN PRINTOUT
ISPHI	INDICATOR FOR CALCULATION OF SPECIAL COMPUTING INTERVAL NEAR TARGET PLANET SPHERE-OF-INFLUENCE
IYR	PRINTOUT CALENDAR YEAR
JJ	COUNTER FOR NUMBER OF ITERATIONS FOR INTERPOLATED SPHERE OF INFLUENCE
LARCA	INDICATOR FOR CALCULATION OF PSUEDO CLOSEST APPROACH
MIN	PRINTOUT CALENDAR MINUTES
NTPI	INDEX OF THE SPACECRAFT VECTORS IN THE F-ARRAY WITH RESPECT TO THE TARGET PLANET
RCM	MAGNITUDE OF POSITION OF VEHICLE RELATIVE TARGET PLANET AT CLOSEST APPROACH
RCM1	PREVIOUS RADIUS OF VEHICLE RELATIVE TO TARGET PLANET
RCM2	PRESENT RADIUS OF VEHICLE RELATIVE TO TARGET PLANET
RDT	NOT USED
RTEMP	SPACECRAFT POSITION AT INTERPOLATED CLOSEST APPROACH IN THE TARGET PLANET EQUATORIAL SYSTEM
SEC	PRINTOUT CALENDAR SECONDS
TIMCR	TIME INCREMENT USED FOR INTERPOLATED CLOSEST APPROACH
TIMIN	TOTAL TIME USED IN ONE INTEGRATED TRAJECTORY
TIM1	CLOCK TIME AT BEGINNING OF TRAJECTORY
TIM2	CLOCK TIME AT END OF TRAJECTORY

TMU	GRAVITATIONAL CONSTANT OF TARGET PLANET (KM**3/SEC**2)
TP	INTERMEDIATE VARIABLE FOR CALCULATION OF SPECIAL COMPUTING INTERVAL NEAR SPHERE-OF-INFLUENCE OF TARGET PLANET
TTG	GRAVITATIONAL CONSTANT OF TARGET PLANET (KM**3/SEC**2)
VCA	VELOCITY MAGNITUDE WITH RESPECT TO TARGET PLANET AT INTERPOLATED CLOSEST APPROACH
VCM	MAGNITUDE OF VELOCITY OF VEHICLE RELATIVE TO TARGET PLANET AT CLOSEST APPROACH BEFORE INTERPOLATION
VQ	EQUATORIAL SPACECRAFT VELOCITY RELATIVE TO TARGET PLANET AT CLOSEST APPROACH BEFORE INTERPOLATION
VTEMP	SPACECRAFT VELOCITY AT INTERPOLATED CLOSEST APPROACH IN THE TARGET PLANET EQUATORIAL SYSTEM
XI	UNINTERPOLATED EQUATORIAL INCLINATION FOR PRINTOUT PURPOSES
XMAG	INTERMEDIATE VARIABLE FOR CALCULATION OF XI
XN	VECTOR NORMAL TO TRAJECTORY PLANE IN TARGET PLANET EQUATORIAL SYSTEM FOR CALCULATION OF XI
XQ	EQUATORIAL SPACECRAFT POSITION RELATIVE TO TARGET PLANET AT UNINTERPOLATED CLOSEST APPROACH
ZM	INTERMEDIATE VARIABLE IN CALCULATION OF INTERPOLATED CLOSEST APPROACH INCLINATION
ZTEMP	VECTOR NORMAL TO TRAJECTORY PLANE FOR CALCULATION OF INTERPOLATED CLOSEST APPROACH INCLINATION

COMMON COMPUTED/USED:	CAINC	DC	DSI	ICL	INCMNT
	INCMT	ISPH	ITRAT	KOUNT	NBODYI
	NO	RCA	RC	RSI	TIMINT
	VSI	V			
COMMON COMPUTED:	DELTH	INCPR	RE	RTP	RVS

VMU

COMMON USED:

ALNGTH	BDR	BDT	B	DELTP
EM7	EM8	F	HALF	ICL2
IEPHEM	INPR	IPRINT	NBOD	NB
NTP	ONE	PLANET	PMASS	RADIUS
RAD	SPHERE	TM	TWO	ZERO

VMP Analysis

VMP provides the logic to integrate an N-body trajectory from an initial spacecraft state ($\bar{r}_S$, v_S) at time t_B to one of the following stopping conditions.

1. Target planet sphere of influence (SOI) is reached (ISP2 $\neq$ 0).

2. The closest approach to the target planet has been reached (ICL2 = 1).

3. The preset final trajectory time t_F has been exceeded.

The integration logic is controlled by ITRAT

ITRAT = 1 First pass through computation cycle (including ephemeris computation).

2 Second pass through computation cycle (excluding ephemeris).

3 Initialization flag.

To start the integration, appropriate variables are initialized (PRINTZ) and ITRAT is set equal to 3. The state of all gravitational bodies at t_B are found (ORB, EPHEM). The initial virtual mass position $\bar{r}_{V_B}$, velocity $\bar{v}_{V_B}$, magnitude μ_{V_B} and magnitude rate $\dot{\mu}_{V_B}$ are found by VMASS. Virtual mass dependent values are then initilized

$$\mu_{V_{AVE}} = \mu_{V_E} = \mu_{V_B} \tag{1}$$

$$\dot{\bar{r}}_{V_{AVE}} = \dot{\bar{r}}_{V_B} \tag{2}$$

$$\bar{r}_{VS_E} = \bar{r}_{VS_B} \tag{3}$$

$$\dot{\bar{r}}_{VS_E} = \dot{\bar{r}}_{VS_B} \tag{4}$$

$$(\Delta t)^2 = 1 \tag{5}$$

$$\text{ISPH1} = 0 \tag{6}$$

At this point the standard integration routine is entered by calling VECTOR.

In the standard integration routine, a new increment is initiated by calling ESTMT which:

1. Initializes all appropriate variables at the beginning of the increment (subscript B) to equal their values at the end of the previous increment.

2. Computes a Δt for the increment based on a modified true anomaly passage.

3. Computes the time at the end of the increment t_E.

4. Estimates the final (subscript E) position $\bar{r}_{V_E}$ and magnitude μ_{V_E} of the virtual mass.

Based on these estimates, the average magnitude and velocity of the virtual mass is computed

$$\mu_{V_{AVE}} = 1/2\ (\mu_{V_B} + \mu_{V_E}) \tag{7}$$

$$\bar{v}_{V_{AVE}} = (\bar{r}_{V_E} - \bar{r}_{V_B})/\Delta t \tag{8}$$

Subroutine VECTOR then computes the orbit relative to the virtual mass based on these estimates. It also refines the estimate of the spacecraft final state ($\bar{r}_{S_E}$, $\bar{v}_{S_E}$). ORB and EPHEM are called to determine the state at t_E of all gravitational bodies being considered. The virtual mass position $\bar{r}_{V_E}$, velocity $\bar{v}_{V_E}$, magnitude μ_{V_E} and magnitude rate $\dot{\mu}_{V_E}$ are determined by VMASS.

Using these refined values, the virtual mass average magnitude $\mu_{V_{AVE}}$ and velocity $\bar{v}_{V_{AVE}}$ are recomputed using equations (7) and (8). At this point a second pass is made through VECTOR to compute the spacecraft final state ($\bar{r}_{S_E}$, $\bar{v}_{S_E}$) which will be used in all subsequent calculations. VMASS is again called to make a final determination of the virtual mass

position, velocity, magnitude and magnitude rate at the end of the increment.

The virtual mass average accelerations are then computed

$$\ddot{\mu}_{V_{AVE}} = \left[\mu_{V_E} - \mu_{V_B} - \dot{\mu}_{V_B}\ (\Delta t)\right] / (\Delta t)^2 \qquad (9)$$

$$\dot{\bar{v}}_{V_{AVE}} = \left[\bar{r}_{V_E} - \bar{r}_{V_B} - \bar{v}_{V_B}\ (\Delta t)\right] / (\Delta t)^2 \qquad (10)$$

These values are subsequently used by ESTMT to estimate the final position $\bar{r}_{V_E}$ and magnitude μ_{V_E} of the virtual mass for the next increment.

Tests are now made to determine whether the target planet SOI has been pierced. If it has the interpolated state at the SOI is found using the following iterative routine

$$\delta r = r_{SOI} - \left| \bar{r}_{ST}^{(n)} \right| \qquad (11)$$

where r_{SOI} = radius of the target planet SOI

$\bar{r}_{ST}^{(n)}$ = position of spacecraft WRT target planet at n-th iteration

$$\delta t^{(n)} = \frac{\delta r \left| \bar{r}_{ST}^{(n)} \right|}{\bar{r}_{ST}^{(n)} \cdot \bar{v}_{ST}^{(0)}} \qquad (12)$$

$$\bar{r}_{ST}^{(n+1)} = \bar{r}_{ST}^{(n)} + \bar{v}_{ST}^{(0)}\ \delta t^{(n)} \qquad (13)$$

$$t_{SI} = t_E^{(0)} \qquad (14)$$

The interpolated state at the SOI is then used by IMPACT to compute B·T and B·R.

If trajectory data is to be printed at this point, the orbit inclination (assuming a hyperbolic orbit about the planet) is computed by first determining the "Kepler vector"

$$\bar{k} = \bar{r}_{ST} \times \bar{v}_{ST} \qquad (15)$$

in planetocentric equatorial coordinates. Then

$$\cos i = \frac{k_z}{|\hat{k}|} \tag{16}$$

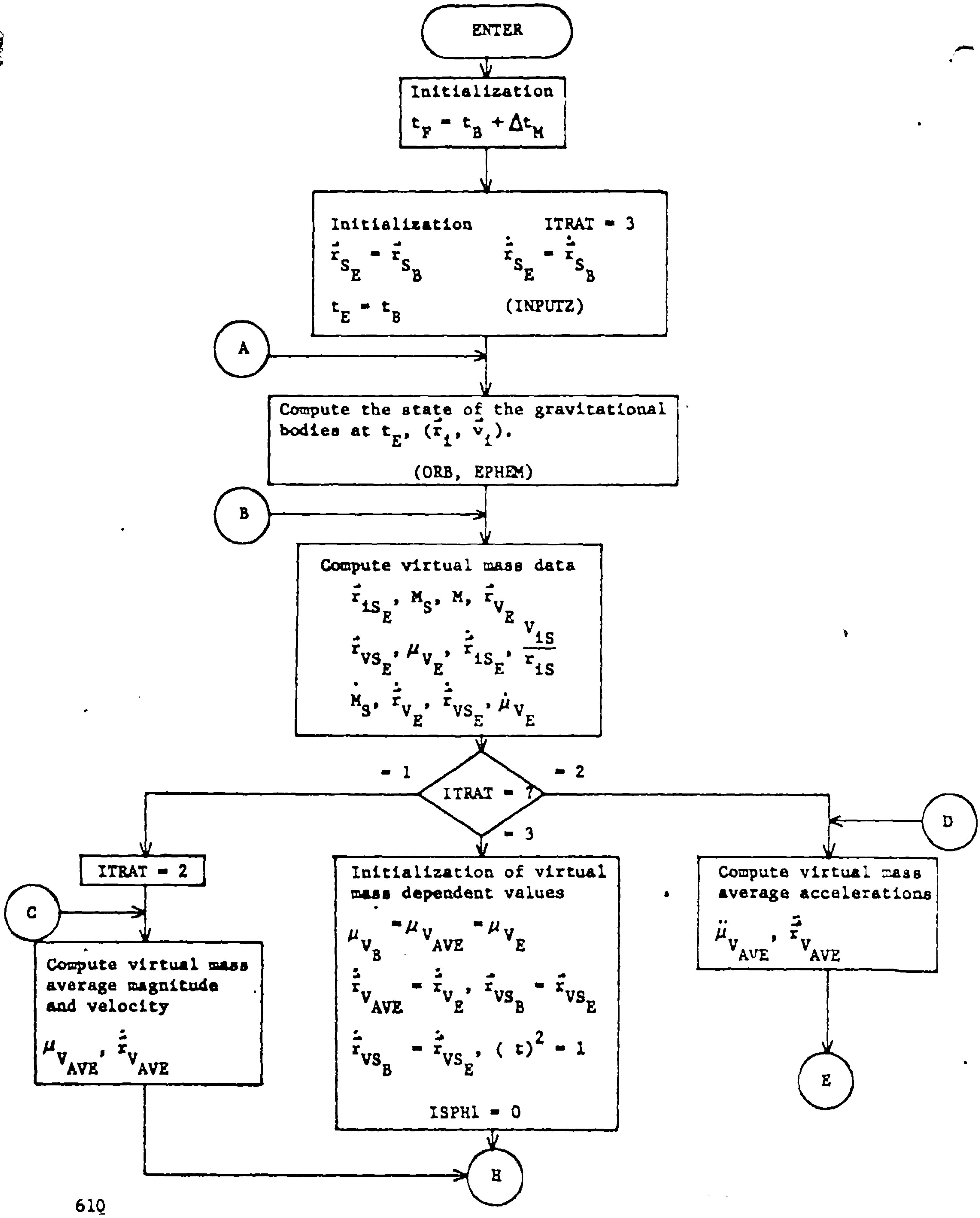
where i = orbit inclination

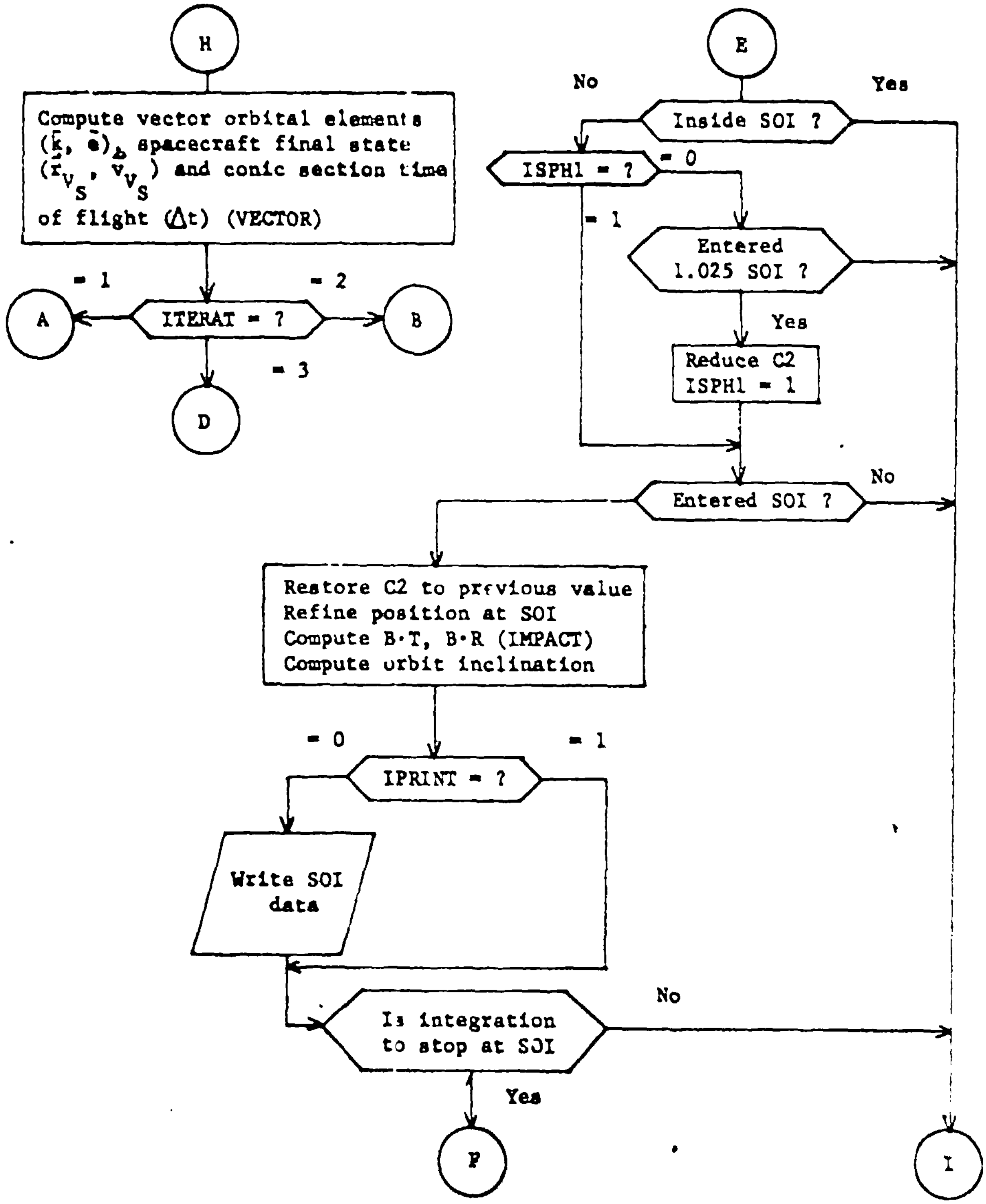
k_z = component of $\hat{k}$ normal to planet equatorial plane

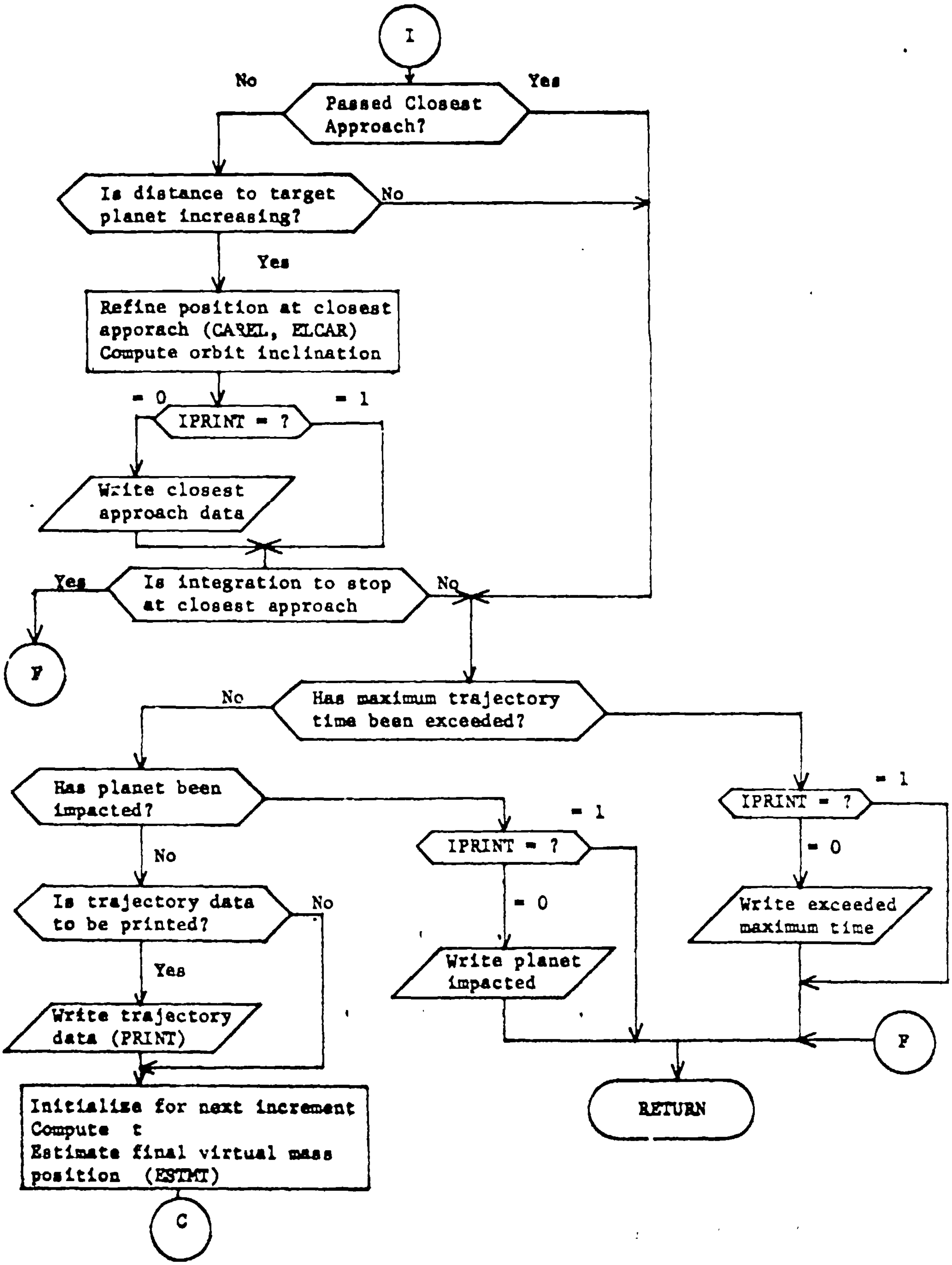
Tests are now made to determine if the spacecraft has reached a closest approach to the target planet. If it has, the interpolated state at closest approach ($\bar{r}_{CA}$, v_{CA}) is computed by calling CAREL with the spacecraft state just following closest approach. CAREL returns the element of the near planet conic. ELCAR is then called with these conic element and returns the interpolated state at closest approach.

If the spacecraft is not within 10 SOI of the target planet, print out of closest approach data may occur; however, integration continues.

The final tests before starting a new integration increment determines if the maximum trajectory time t_p has been exceeded or a planet has been impacted. If these tests are passed, a new integration cycle is initiated by calling ESTMT.


ENTER
Initialization
$t_F = t_B + \Delta t_M$
Initialization ITRAT = 3
$\vec{r}_{S_E} = \vec{r}_{S_B}$ $\dot{\vec{r}}_{S_E} = \dot{\vec{r}}_{S_B}$
$t_E = t_B$ (INPUTZ)
A
Compute the state of the gravitational bodies at t_E, $(\vec{r}_i, \vec{v}_i)$.
(ORB, EPHEM)
B
Compute virtual mass data
$\vec{r}_{iS_E}$, M_S, M, $\vec{r}_{V_E}$
$\vec{r}_{VS_E}$, μ_{V_E}, $\dot{\vec{r}}_{iS_E}$, $\frac{V_{iS}}{r_{iS}}$
$\dot{M}_S$, $\dot{\vec{r}}_{V_E}$, $\dot{\vec{r}}_{VS_E}$, $\dot{\mu}_{V_E}$
ITRAT = ?
= 1
= 2
= 3
ITRAT = 2
C
Compute virtual mass average magnitude and velocity
$\mu_{V_{AVE}}$, $\dot{\vec{r}}_{V_{AVE}}$
Initialization of virtual mass dependent values
$\mu_{V_B} = \mu_{V_{AVE}} = \mu_{V_E}$
$\dot{\vec{r}}_{V_{AVE}} = \dot{\vec{r}}_{V_E}$, $\vec{r}_{VS_B} = \vec{r}_{VS_E}$
$\dot{\vec{r}}_{VS_B} = \dot{\vec{r}}_{VS_E}$, $(t)^2 = 1$
ISPH1 = 0
D
Compute virtual mass average accelerations
$\ddot{\mu}_{V_{AVE}}$, $\ddot{\vec{r}}_{V_{AVE}}$
E
H


H
Compute vector orbital elements
$(\bar{k}, \bar{e})_b$ spacecraft final state
$(\bar{r}_{V_S}, \bar{v}_{V_S})$ and conic section time
of flight (Δt) (VECTOR)
= 1
= 2
ITERAT = ?
A
B
= 3
D
E
No
Yes
Inside SOI ?
ISPH1 = ?
= 0
= 1
Entered
1.025 SOI ?
Yes
Reduce C2
ISPH1 = 1
No
Entered SOI ?
Restore C2 to previous value
Refine position at SOI
Compute B·T, B·R (IMPACT)
Compute orbit inclination
= 0
= 1
IPRINT = ?
Write SOI
data
No
Is integration
to stop at SOI
Yes
F
I


I
No
Yes
Passed Closest Approach?
Is distance to target planet increasing?
No
Yes
Refine position at closest apporach (CAREL, ELCAR)
Compute orbit inclination
= 0
= 1
IPRINT = ?
Write closest approach data
Yes
Is integration to stop at closest approach
No
F
No
Has maximum trajectory time been exceeded?
Has planet been impacted?
= 1
IPRINT = ?
= 1
IPRINT = ?
= 0
No
= 0
Is trajectory data to be printed?
No
Write exceeded maximum time
Yes
Write planet impacted
Write trajectory data (PRINT)
F
Initialize for next increment
Compute t
Estimate final virtual mass position (ESTMT)
RETURN
C

SUBROUTINE   ZERIT

PURPOSE: TO COMPUTE THE COMPUTATION OF THE ZERO ITERATE VALUES OF TIME, POSITION VECTOR, AND VELOCITY VECTOR.

CALLING SEQUENCE: CALL ZERIT

SUBROUTINES SUPPORTED: PRELIM   GIDANS

SUBROUTINES REQUIRED: HELIO   LUNA

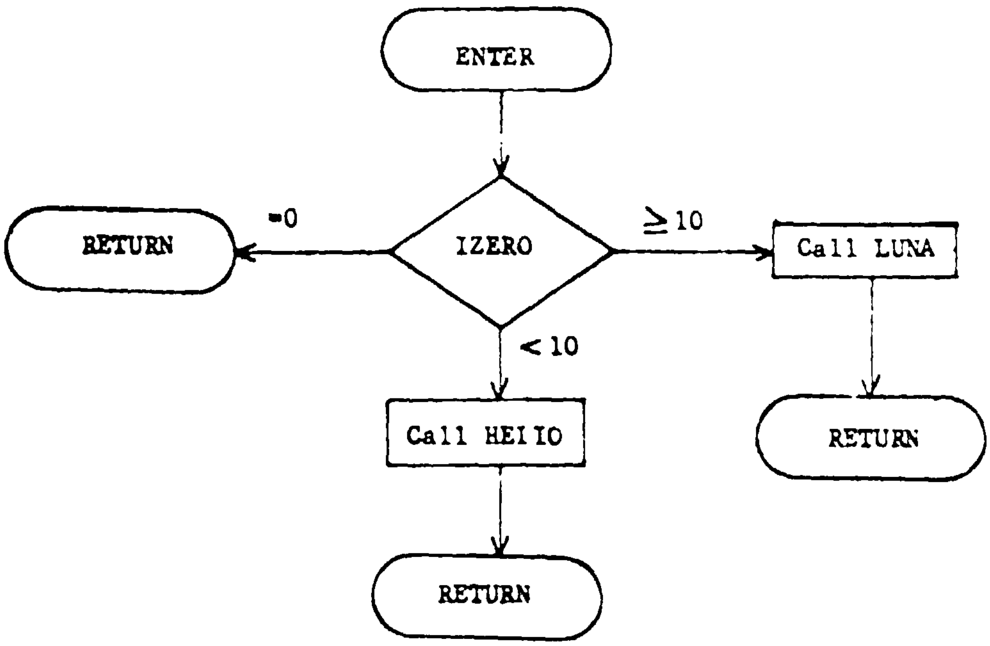
COMMON USED: IZERO   LTARG

ZERIT Analysis

ZERIT is the executive subroutine handling the computation of the zero iterate values of time, position vector, and velocity vector.

The flag IZERO controls the operation of ZERIT. If IZERO = 0, no zero iterate computation is needed and so ZERIT is exited. If IZERO < 10, the zero iterate is to be computed for a interplanetary trajectory so HELIO is called before returning. If IZERO ≥ 10, the zero iterate is to be computed for a lunar trajectory so LUNA is called for that computation.

ZERIT Flow Chart



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